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AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST CH-47C HELICOPTER (CHINOOK)

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PERFORMANCE

FINAL REPORT

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST

CH-47C HELICOPTER (CHINOOK)

PERFORMANCE.

9 FINAL REPORT, 29 Apr 69-21 Aug 70,

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ABSTRACT

The CH-47C was flight tested to obtain detailed performance data and to verify compliance of the aircraft with the manufacturer's detail sepcification and applicable military specifications. The test results show that the helicopter exceeded all performance guarantees and complied with all specifications against which it was tested.except airspeed position errors. The inaccuracy of the engine torquemeter system and high engine compartment vibration levels were the only two deficiencies found. Seven shortcomings were noted for which correction is desirable: (1) objectionable cockpit vibration levels which limit maximum level-flight airspeed, (2) moderate pilot effort required to maintain optimum climb airspeeds, (3) 3/rev airspeed indicator needle oscillations at high power settings, (4) engine torque mismatch resulting from adjusting rotor speed, (5) use of landing gear power steering control may be lost at gross weights below 30,000 pounds, (6) objectionable cargo compartment vibration, and (7) objectionable noise levels in the cockpit. The small airspeed system position error associated with changes in vertical speed represent a marked improvement over the systems in the CH-47A and the CH-47B. The greatly improved hover capability and excellent climb performance enhance the operational suitability of the helicopter. The use of a cruise guide indicator to display inflight loads on the aft dynamic components of the flight control system is excellent and should be incorporated in future designs. The performance characteristics of the helicopter are satisfactory for operational use.

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DISTRIBUTION

INTRODUCTION

BACKGROUND

1. Experience with the CH-47A/B helicopter in Vietnam has verified the importance of improving both payload and speed capability at high density altitudes (H_D). These increased capabilities would provide for better combat effectiveness and utilization of the aircraft.

2. The product improvement program (ref 1, app I) defines a two-step program to incorporate performance, stability, and vibration-level improvements in production CH-47 helicopters. Aircraft configured for step-one modifications have been identified as configuration 1A and are designated CH-47B. The second step in the product improvement program provides for the incorporation of increased shaft horsepower (shp) and necessary modification to accomodate the higher power for a further increase in payload capability. Aircraft configured for step-two modifications have been identified as configuration II and are designated CH-47C.

3. The test directive issued by the US Army Test and Evaluation Command (TECOM) (ref 2, app I) directed the US Army Aviation Systems Test Activity (USAASTA) to participate in the CH-47 product improvement program. This participation included the conduct of tests on the production configuration CH-47C to acquire detailed performance, vibration, and stability and control information. The test directive (ref 3) issued by the US Army Aviation Systems Command (AVSCOM) provided additional guidance and forwarded agreed changes to the test plan which were incorporated (ref 4). The CH-47C airworthiness and flight characteristics (A&FC) test program was divided into two phases: performance, and stability and control. This report discusses the performance phase of the program.

TEST OBJECTIVES

4. The objectives of the A&FC performance tests were to obtain and compile performance data on a production CH-47C helicopter for incorporation into technical manuals. Tests were conducted to determine the following:

a. Degree of conformance with the detail specification (ref 5, app I).

b. Degree of conformance with the T55-L-11 engine model specification (ref 6, app I).

c. Conformance with the portion of MIL-I-6115A (ref 7, app I) applicable to airspeed and altimeter systems.

DESCRIPTION

5. The CH-47C helicopter is manufactured by the Vertol Division of The Boeing Company (Boeing-Vertol). It is a dual-engine, turbine-powered, tandem-rotor aircraft designed to provide air transportation for cargo, troops, and weapons within the combat area. The helicopter is powered by two Lycoming T55-L-11 turboshaft engines mounted in separate nacelles on the aft portion of the fuselage. The engines drive two three-bladed rotors in tandem through a combining transmission, drive shafting, and reduction transmissions. A gas turbine hydraulic auxiliary power unit (APU) drives the aft transmission accessory gearbox to provide hydraulic and electrical power for engine starting and other ground operations when the rotors are not turning. Two pods, containing three fuel tanks each, are located on either side of the fuselage. The helicopter is equipped with four nonretractable landing gear. An entrance door is located at the forward right side of the cabin fuselage section. A hydraulically powered loading ramp is located at the rear of the cargo compartment. Side-by-side seating arrangements are provided for the pilots. All tests, except those noted below, were conducted with the cargo mirror and engine inlet screens removed, and also with the cargo ramp and lip, all doors, windows, and cargo hook hatch closed. Landing and takeoff tests were conducted with the engine inlet screens installed and cargo hook hatch removed. Hover tests were conducted with the cargo hook hatch removed. The physical characteristics of the CH-47C helicopter are presented in appendix II. A detailed explanation of the engineering changes which have been incorporated in the CH-47C can be found in the product improvement report (ref 1, app 1). The significant changes from the CH-47B are as follows:

a. T55-L-11 engines rated at 3,750 maximum shp at sea-level (SL), standard-day conditions (Engineering Change Proposal (ECP) 449)).

b. Uprated engine transmission and combining transmission (ECP 446 and ECP 447).

c. Increased torque load carrying capability of the forward transmission (ECP 435).

d. Increased torque load carrying capability of the aft transmission (ECP 436).

e. Strengthened synchronizing shaft adapters, engine drive shaft, and engine drive shaft adapter (ECP 448).

f. Increased capacity of the lag dampers (ECP 451).

g. Increased fuel capacity (ECP 553).

h. Increased wall thickness of the aft rotor shaft (ECP 402).

i. Installation of automatically tuned vibration absorbers (ECP 554).

j. Revised forward cyclic trim (ECP 598).

k. Incorporation of balance springs to collective and directional controls (ECP 610).

l. Installation of pitch stability augmentation (PSA) system (ECP 611R1).

m. Reduced lateral control sensitivity and added limited roll attitude retention (ECP 620).

n. Revised aft pylon vibration absorber tuning (ECP 574).

o. Revised aft upper controls (ECP 585).

p. Installed flight-load indicating system (ECP 556). (Installed in the helicopter for evaluation of the cruise guide indicator (CGI) system and as an aid in conducting the tests. The system is not presently incorporated in production aircraft.)

SCOPE OF TEST

6. During the test program, 99 flights were conducted for a total of 151.6 hours, of which 95.9 hours were productive. Of the nonproductive time, 32.9 hours were used for ferrying the aircraft to the various test sites, 10.3 hours were used for functional check flights and instrumentation checks, and the remaining flight hours were used for flying to the local test areas and returning. Testing was conducted from 13 October 1969 to 21 August 1970 in California at the US Naval Air Facility, El Centro (-43 feet), Coyote Flats (9,500 feet), Edwards Air Force Base (2,302 feet), and Shafter (420 feet), and in Canada at the Canadian Forces Base Cold Lake, Alberta (1,774 feet).

7. The CH-47C was evaluated with respect to its mission as a transport helicopter as defined in the detail specification (ref 5, app 1). Performance results were compared to the guarantees set forth in the detail specification and are presented in paragraph 14.

8. The normal operating limitations listed in reference 8, appendix 1, as modified by the test directive (ref 4), were observed during all tests.

METHODS OF TEST

9. Test methods and data reduction procedures used in these tests are proven engineering flight test techniques and are described briefly in appendix III. A more detailed discussion is contained in references 9 through 13, appendix 1.

10. Data were recorded on a photopanel and oscillograph utilizing calibrated sensitive instruments. A detailed list of test helicopter instrumentation is included as appendix IV.

11. Flying qualities characteristics, where appropriate, were evaluated during performance tests. The Handling Qualities Rating Scale (HQRS) was used to augment qualitative comments and is presented as appendix V.

CHRONOLOGY

12. The chronology of the CH-47C A&FC performance test program is as follows:

Test request received	29	April	1969
Aircraft received	12	May	1969
Engineering flight tests started	13	October	1969
Engineering flight tests completed	21	August	1970
Report sent to AVSCOM for author review	11	January	1971
Report returned to USAASTA	22	February	1971
Advance copy of report submitted		September	1971

RESULTS AND DISCUSSION

GENERAL

13. Flights were conducted on a production model CH-47C to obtain detailed performance data for use in determining compliance with the detail specification (ref 1, app I) and applicable military specifications. The data also provide information for use in technical manuals and other publications. The CH-47C exceeded all contract performance guarantees. A summary of the performance guarantee compliance is presented in table 1. Torquemeter system inaccuracy and high engine compartment vibration levels were the only deficiencies that affected mission accomplishment. There were seven shortcomings for which correction is desirable: (1) objectionable cockpit vibration levels which limit maximum level-flight airspeed, (2) moderate pilot effort required to maintain optimum climb airspeeds, (3) airspeed indicator needle oscillations (3/rev) at high power settings, (4) engine torque mismatch which results from adjusting rotor speed, (5) the use of landing gear power steering control at gross weights below 30,000 pounds, (6) objectionable cargo compartment vibration, and (7) objectionable noise levels. The 6,000-pound (15-percent) increase in maximum gross weight (grwt) and the 4,600-pound (22-percent) increase in payload capability of the CH-47C over that of the CH-47B represents a significant increase in the operational effectiveness of the helicopter. The airspeed at which unacceptable cockpit vibration levels occur on the CH-47C has been increased approximately 20 knots (17 percent) above that of the CH-47B at the light gross weights. The reduction in airspeed system position error associated with changes in vertical speed represents a marked improvement over the CH-47A and CH-47B. The greatly improved hover capabilities and the excellent climb performance enhance the operational capability of the helicopter. The use of a cruise guide indicator to show inflight loads on the aft dynamic components is excellent and should be incorporated in present and future CH-47C helicopters.

14. Table 1 lists guarantees based upon the specified mission gross weights and rotor speeds where applicable. Performance guarantees are quoted for an aircraft configured for an internal cargo mission (no outside mirror, no troop seats, and without inlet screens or separators) at a 245-rpm rotor speed, unless stated otherwise. Guarantee compliance was demonstrated in accordance with Boeing-Vertol report number 114-TN-601, revision A (ref 4, app I), as approved by the procuring activity.

Table 1. Performance Guarantee Summary.

Condition	Unit	Guarantee	Test Results
Mission I[1] payload, outbound	lb	12,000	[2]12,000
Mission I[1] payload, inbound	lb	6,000	[2]6,000
Mission I[1] radius of action	NM[3]	100	[2]100
Mission I[1] service ceiling, single engine, military power (MP)	ft	4,000	[4]6,500
Mission I[1] OGE hover capability, 95°F day	ft	6,000	[5]6,680
Mission II[6] maximum cruise speed, SL, standard day, normal power (NP)	KTAS[7]	155	158
Mission III[8] OGE hover capability, SL, standard day	lb	43,000	44,450

[1]Mission I. The helicopter shall be capable of hovering at 6,000 feet for 10 minutes at 95°F, OGE, at gross weight required for accomplishment of Mission I (guaranteed). The Mission I gross weight includes an outbound payload of 12,000 pounds, return payload of 6,000 pounds, and fuel for a radius of 100 NM.
[2]Value fixed to determine the Mission I gross weight (ref 5, app I).
[3]Nautical mile.
[4]Results calculated from level flight performance.
[5]Results calculated from generalized hover performance.
[6]Mission II. The aircraft shall possess the ability to cruise at 155 knots at its design gross weight of 33,000 pounds.
[7]Knots true airspeed.
[8]Mission III. The helicopter shall be capable of hovering OGE at SL, standard-day, maximum power conditions at a gross weight of 43,000 pounds (guaranteed).

HOVER PERFORMANCE

15. The objectives of the hover performance tests were to determine the in-ground-effect (IGE) and out-of-ground-effect (OGE) power required as a function of aircraft gross weight, density altitude, and rotor speed; and to determine detail specification compliance. Tests were conducted using the tethered flight method. Hover data were gathered at field elevations of approximately SL and 9,500 feet. The SL data contained the majority of low referred rotor speed data and included performance from 217 to 248 rpm, while the 9,500-foot density altitude data included referred rotor speeds from 225 to 252 rpm. Data were obtained at hover heights of 5, 10, 20, 50, and 150 feet (referenced to the bottom of the right rear tire). The test technique and data analysis methods are described in paragraphs 11 through 18, appendix III.

16. The IGE hover capability at a 10-foot wheel height for a standard day and a 95°F day is presented in figure 1, appendix VI. At the alternate design gross weight (46,000 pounds), a 10-foot wheel height, and a 245-rpm rotor speed, the aircraft can hover at 8,250 feet on a standard day.

17. The OGE hover performance in figure 3, appendix VI, shows that at 37,474 pounds the CH-47C helicopter can hover OGE at a 6,680-foot pressure altitude on a 95-degree day at a 245-rpm rotor speed. This exceeds the Mission I guarantee by 680 feet (11.3 percent). The standard-day, SL, OGE hover capability at a 245-rpm rotor speed is 44,450 pounds, which exceeds the Mission III hover capability guarantee by 1,450 pounds (3.4 percent). A comparison between the useful load capabilities of the CH-47B and the CH-47C is shown in table 2. The hover performance of the CH-47C is greatly improved over that of the CH-47B. This increased payload capability enhances the operational capability of the helicopter.

Table 2. Out-of-Ground-Effect Useful Load Comparison Summary.

Aircraft	Empty Weight	Useful Load[1]	Useful Load[2]	Rotor Speed (rpm)
CH-47B	20,068	19,900	19,250	230
CH-47C	20,213	24,237	23,867	245

[1]SL, standard-day conditions.
[2]SL, 95°F-day conditions.

TAKEOFF PERFORMANCE

18. The objective of the takeoff performance test was to determine the takeoff distance required to clear a 100-foot obstacle. The tests were conducted at a field elevation of 9,500 feet mean sea level (MSL) and at gross weights ranging from 39,000 to 43,000 pounds at a mid center of gravity (cg). Rotor speed was maintained at 245 rpm.

19. The level acceleration from a hover to a constant climbout airspeed technique was used. Takeoffs were initiated from a 10-foot hover, when sufficient power was available, with maximum power applied at the initiation of forward motion. When sufficient power was not available to hover at 10 feet, takeoffs were initiated at the hover height obtainable with maximum power. Takeoffs at a hover height of less than 8 feet were not attempted. During acceleration, the pilot attempted to maintain level flight; however, the flight path varied from 5 to 10 feet above the ground. Rotation to a climbout attitude was initiated approximately 5 knots below the target airspeed and maintained until the obstacle was cleared. A Fairchild Flight Analyzer was used to record ground speed and horizontal distance required to clear a 100-foot obstacle. The data reduction method is described in paragraphs 19 through 21, appendix III, and the results are presented in figures 13 through 18, appendix VI.

20. Takeoff tests conducted at gross weights where OGE hover could not be attained resulted in the aircraft settling toward the ground when rotation to a climb attitude was initiated prematurely. This occurred when the power available was less than the power required for OGE level flight at the climbout airspeed. At higher airspeeds, approximately 50 knots true airspeed (KTAS), the aircraft could maintain a positive rate of climb beyond the 100-foot obstacle. When the maximum hover height was 10 feet or less, the acceleration prior to rotation demanded considerable pilot effort and technique to prevent the aircraft from contacting the ground. When power was insufficient to hover higher than 10 feet, the horizontal distance required to clear the 100-foot obstacle varied from 1,775 feet at 50 KTAS to 2,395 feet at 70 KTAS (fig. 14, app VI). The technique of accelerating to the higher airspeed prior to rotation (approximately 70 knots indicated airspeed (KIAS)), providing space is available, provides a higher rate of climb before and after the 100-foot obstacle is cleared and would also provide for a higher margin of safety in the event of an engine failure. The level acceleration takeoff technique should not be used when power available is insufficient to hover higher than 10 feet.

21. During the takeoff tests, the pilot experienced difficulty in maintaining a precise rotor speed during the level acceleration to climb attitude. To alleviate this problem, the copilot monitored the power parameters and control rotor speed by manipulation of the beeper and thrust control rod. This procedure allowed the pilot to concentrate on controlling the aircraft during takeoff. The operator's manual should reflect the technique of using the copilot to monitor and control the power parameters during takeoff where power available is insufficient to acccomplish vertical takeoffs.

FORWARD FLIGHT CLIMB PERFORMANCE

22. The objective of these tests was to determine the maximum climb airspeed schedule and rates of climb up to service or envelope ceiling, whichever was reached first. The climb airspeed schedule was then compared with that calculated from the level-flight performance data.

23. Sawtooth climbs were conducted to determine the power and gross weight correction factors at referred gross weights of 26,540, 45,620, and 51,530 pounds at various power settings and density altitudes. The results of these climbs are presented in figures 23 and 24, appendix VI. The tests show that the power correction factor (K_p) remained essentially constant throughout the gross weights tested.

24. Continuous climbs were conducted for both dual-engine and single-engine operation. Single-engine climbs at military rated power (MRP) to service ceiling were conducted at a rotor speed of 230 rpm at 37,365 pounds, the approximate Mission I gross weight. Dual-engine climbs at normal rated power (NRP) to the envelope limit altitude of 15,000 feet were conducted at gross weights of 26,235 and 33,355 pounds at a rotor speed of 235 rpm. At 46,295 pounds, a climb to service ceiling was conducted at 245 rpm. All climb data were adjusted for power, rpm, gross weight, and air density variations as defined in appendix III.

25. The contractor climb airspeed schedules were within ±2 knots of the best rate-of-climb airspeed obtained from level flight.

26. Cockpit vibration levels were evaluated during the climb tests and are satisfactory for operational use.

27. The single-engine MRP climb performance (derived from the level-flight generalized power-required curves and power-available curves as specified in ref 6, app I) was used to determine compliance with the Mission I guarantees. These calculated results show a single-engine service ceiling of 6,600 feet, which exceeds the guarantee by 2,600 feet (65 percent). The single-engine service ceiling at Mission I gross weight is satisfactory for operational use.

28. The dual-engine climb performance of the CH-47C is presented in figures 20 through 22, appendix VI. The aircraft is limited to an altitude of 15,000 feet due to possible cavitation of the flight control hydraulic boost pumps. Table 3 presents the rate of climb at altitude ceiling. The CH-47C has demonstrated that the altitude ceiling can be achieved throughout the allowable gross weight range. Dual-engine climb performance is satisfactory for operational use. At altitudes below 5,000 feet, a pitch oscillation was encountered which required the pilot to make numerous longitudinal control corrections in order to fly the climb airspeed schedule (HQRS 4). Above this altitude, the pilot could fly the climb schedule with a minimum of control inputs (HQRS 2). Increasing the climb airspeed schedule by approximately 10 knots would decrease pilot effort at low altitudes with a degradation in climb performance of approximately 100 feet per minute (ft/min).

This degradation is compensated by reduced pilot effort during climbs. The climb airspeed should be increased to 80 KIAS for night operation, sling load operations, and instrument flight, and should be used any time maximum climb performance is not required. The forward flight dual-engine climb performance of the CH-47C helicopter enhances the mission capability of the aircraft.

Table 3. Rate of Climb at Altitude Ceiling.[1]

Gross Weight (lb)	Rotor Speed (rpm)	Pressure Altitude (ft)	Rate of Climb ft/min
26,235	235	15,000	2,300
33,255	235	15,000	1,225
46,795	245	8,000	445

[1]Dual-engine normal rated power (NRP).

LEVEL FLIGHT PERFORMANCE

29. The objective of these tests was to determine the variation of power required with rotor speed, airspeed, and gross weight. From these relationships, specific range, endurance airspeed, maximum airspeed, level-flight engine performance characteristics, and detail specification guarantees were determined.

30. Level-flight performance data were acquired using the constant referred rotor speed ($N_R/\sqrt{\theta}$) and referred gross weight (W/δ) method of test. Data were obtained at constant referred gross weights ranging from 26,060 to 60,520 pounds for constant referred rotor speeds of 225 to 268 rpm. Previous tests conducted on the CH-47 helicopter have shown that cg does not have a significant effect on power required during level flight. Therefore, all tests conducted during this program were conducted at a mid cg.

31. The generalized power-required curves derived from these tests are presented in figures 25 through 41, appendix VI. The computation of Mission I gross weight is presented in table 4. Computation of the fixed useful load for accomplishing Mission I is presented in table 5. The radius-of-action summary plot is presented in figure 42, appendix VI. Range summaries at SL and 5,000 feet are presented in figures 43 and 44.

Table 4. Computation of Mission I Gross Weight.[1]

Item	Weight (lb)
Detail specification Weight Empty Statement	20,420
Troop seats	-169
Engine inlet screen	-38
Mission I empty weight	20,213
Fixed useful load	739
Fuel	4,522
Outbound payload	12,000
Mission I gross weight	37,474
Engine start gross weight	37,474
Warm-up (2 minutes) at NRP	-111
Outbound fuel	-2,005
Landing gross weight	35,358
Average outbound gross weight	36,361
Offload 12,000 pounds, load 6,000 pounds	29,358
Warm-up (2 minutes) at NRP	-111
Inbound fuel	-1,843
Landing gross weight	27,404
Average inbound gross weight	28,325
Unload 6,000 pounds	21,404
Fixed useful load	-739
Empty weight plus fuel	20,665
Ten-percent fuel reserve	-452
Mission I empty weight	20,213
Average specific range outbound:[2] 0.0499 NAMPP[3] Outbound fuel: 2,005 lb Outbound range at average 135 KTAS:[2] 100 NM	
Average specific range inbound:[2] 0.0543 NAMPP Inbound fuel: 1,843 lb Inbound range at average 125 KTAS:[2] 100 NM	

[1]Based on SL, standard-day conditions, T55-L-11 engines installed, bleed air OFF, heater OFF, all windows and doors closed, cargo mirror not installed, and 245 rpm rotor speed.
[2]Average cruise speed at specific range as defined by MIL-C-5011A, specific range for weights shown in computation of Mission gross weight above.
[3]Nautical air miles per pound of fuel.

Table 5. Fixed Useful Load for Accomplishing Mission I.

Item	Weight (lb)
Three crew members at 200 pounds each	600
Unusable fuel	36
Engine oil	53
Cargo tiedown devices	50
Total:	739

32. Figure A shows that the CH-47C exceeded the detail specification Mission II maximum airspeed guarantee of 155 KTAS by 3 knots (1.9 percent). The maximum velocity at altitudes above 5,000 feet was usually limited by cockpit vibration or cruise guide indicator limit. At altitudes below 5,000 feet and gross weights of 33,000 pounds and below, the maximum velocity of the helicopter was generally limited by the transmission torque limit of 1,015 foot-pounds (ft-lb). The never-exceed airspeed (V_{NE}) for the operational envelope could be easily reached prior to achieving the transmission torque limit.

33. On a SL standard day at all operational rpm's, the maximum endurance airspeed is 67 KTAS at a 26,000-pound grwt, increasing to 85 KTAS at a 46,000-pound grwt. The operational endurance airspeed should be established at 80 KIAS if either time or operational conditions do not permit use of the exact endurance speeds published in the performance tables of the operator's manual. The level flight performance of the CH-47C helicopter is satisfactory for operational use.

AUTOROTATIONAL DESCENT PERFORMANCE

34. The objective of these tests was to determine, for various gross weights, the optimum airspeed and rotor speed for minimum rate of descent and maximum glide distance in power-off flight.

35. Autorotational descent performance data were obtained at gross weights of 24,600, 26,130, and 33,270 pounds at an average density altitude of 5,000 feet with the aircraft ballasted to a mid cg. The airspeeds for minimum rate of descent and best glide distance were determined at a rotor speed of 230 rpm, as specified by the operator's manual. The effects of rotor speed on descent performance were investigated at rotor speeds of 214 to 261 rpm (the maximum allowable range).

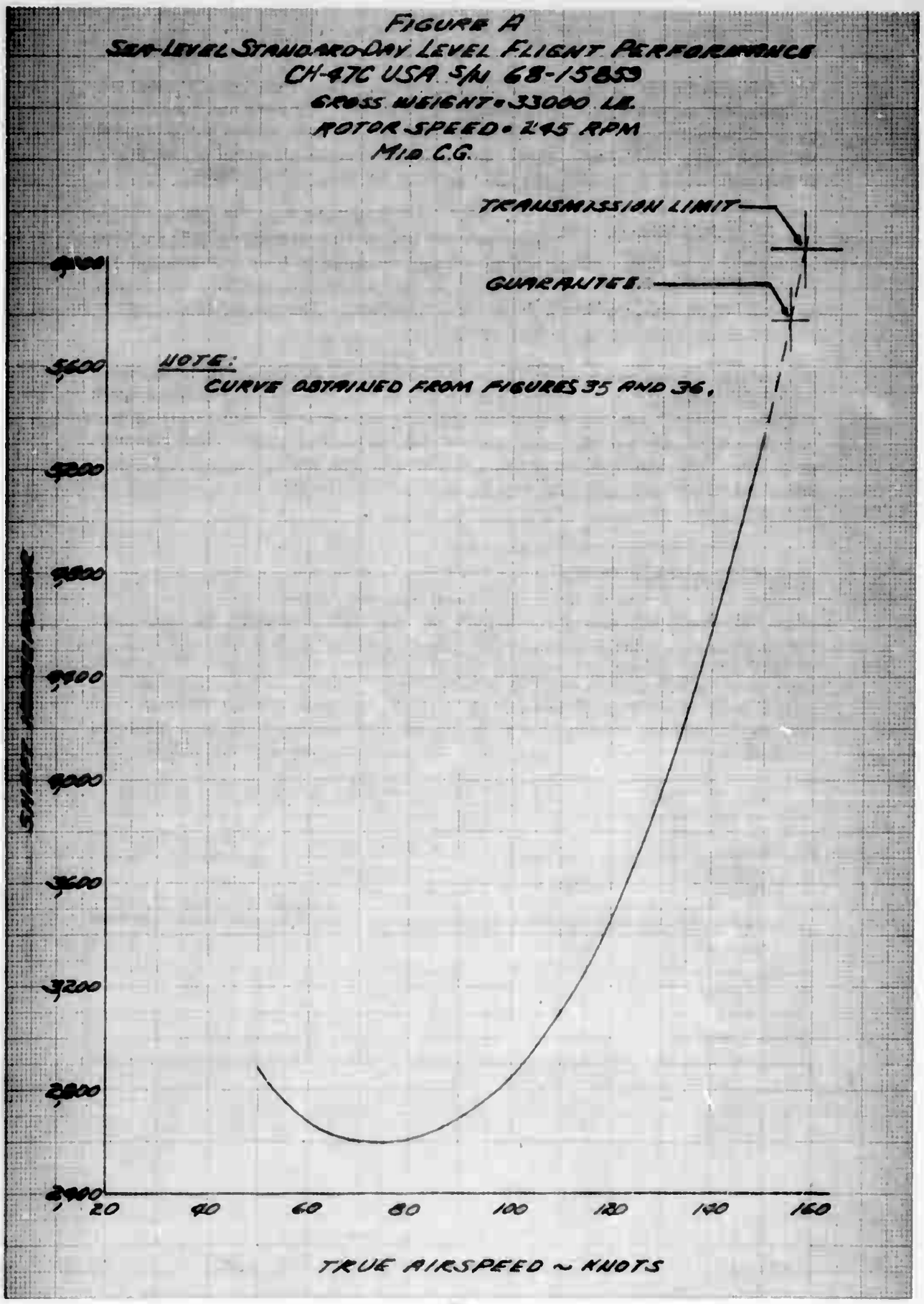
FIGURE A
SEA-LEVEL STANDARD-DAY LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
GROSS WEIGHT = 33000 LB.
ROTOR SPEED = 245 RPM
MID C.G.
TRANSMISSION LIMIT
GUARANTEE
NOTE:
CURVE OBTAINED FROM FIGURES 35 AND 36.
4000
3600
3200
3000
2800
20
40
60
80
100
120
140
160
TRUE AIRSPEED ~ KNOTS

36. The autorotational descent performance derived from these tests is presented in figure 45, appendix VI. The data show that the rate of descent was not affected by gross weight at the weights tested. In addition, the rate of descent decreased with decreasing rotor speed. Future tests should be conducted to define the heavy gross-weight (40,000 pounds and above) autorotational characteristics.

37. Under standard-day conditions at an altitude of 5,000 feet at the test gross weight, the maximum glide distance was achieved at 104 KIAS (112 KTAS). Rate of descent is relatively insensitive to changes in airspeed from the airspeed for minimum rate of descent (80 KTAS). A change in airspeed of ±10 knots results in a rate-of-descent increase of less than 100 ft/min. This characteristic greatly reduces the pilot workload during autorotation.

38. Above a rotor speed of 255 rpm, the cockpit vibration levels increased from moderate to heavy, and the pilot was unable to read the instrument panel gages (HQRS 6). Rotor speed was sensitive and very responsive to small thrust control rod movements. Precise rotor speed control required moderate pilot effort; however, rotor speed can easily be maintained within operating limits. The autorotational descent performance characteristics of the CH-47C helicopter are satisfactory for operational use.

LANDING PERFORMANCE

39. The objectives of these tests were to define an operational technique and the accompanying performance while landing over a 100-foot obstacle at gross weights and/or power conditions which preclude safe vertical descent. The landing tests were conducted at a field elevation of 9,500 feet, in conjunction with and at the same target gross weights as the takeoff performance tests. The tests were limited to internally carried loads. A summary of landing distances is presented in table 6.

Table 6. Summary of Landing Distances Over a 100-Foot Obstacle.

Gross Weight (lb)	Center-of-Gravity Location	Indicated Airspeed (kt)	Distance Required From 100 Feet to Hover (ft)
35,000	Mid	40	875
39,000	Mid	40	1,060
43,000	Mid	40	1,410

40. The landing technique used consisted of executing a shallow approach (5 to 8 degrees) at approximately 40 KIAS. Between 100 and 150 feet above the ground, a gradual deceleration was initiated while maintaining the descent angle. As the aircraft decelerated, power was gradually increased so as to arrive at the hover point with maximum power available and zero ground speed. A distance of from 200 to 300 feet was required to stop the helicopter after a 5- to 10-foot rear wheel height had been achieved. Attempts to stop the helicopter immediately upon reaching a low hover invariably resulted in either ground contact at the termination of the approach or transient increases in power in excess of the transmission limit. The landing performance characteristics of the CH-47C helicopter are satisfactory for operational use.

MISCELLANEOUS

Ground Handling Characteristics

41. To ground taxi using power steering, both pilots are required to perform separate duties. One pilot physically monitors and restricts all control movements while the other pilot operates the brakes and power steering control knob. This prevents either pilot from accomplishing other tasks, such as copying instrument clearances, while the helicopter is being taxied. This inability for one pilot to ground taxi with power steering is undesirable and reduces mission effectiveness.

42. When the helicopter is ground operated at light gross weights (less than 30,000 pounds) with power steering engaged, it is possible for the aft right-hand landing gear to become airborne, which causes a loss of the use of the power steering control. Correction of this shortcoming is desirable for improved mission effectiveness.

43. Ground taxi of the CH-47C with the power steering off can be accomplished with moderate pilot effort using the technique recommended in the operator's manual.

Power Management System and Rotor Speed Control

44. During powered flight, the rpm control of the rotor system is achieved by the two engine beep trim switches located on the thrust control rod. The switch on the left side controls the power output of the #1 (left) engine; the normal engine control switch (#1 and #2) is located on the right side, and controls the power output of the #1 and #2 engines combined. To change rotor speed, the pilot must position these beeper switches either forward (to increase rpm) or aft (to decrease rpm). Once the rotor speed was set and torques were matched by manipulation of these beeper switches, relatively good torque match could be maintained throughout the entire travel of the thrust control rod. Difficulty arose in that rpm varied with collective position, although the #1 and #2 engine torque settings remained matched. This variance necessitated additional beeping of the #1 and #2 switch (right switch) to achieve the desired rpm for the particular

gross weight. When this rpm matching was attempted, major torque mismatching occurred which required further manipulation of both beeper switches. This manipulation detracted from the pilot's and/or copilot's ability to devote attention to other cockpit or flight duties and became especially critical during heavy gross weight sling-load operations (HQRS 5). Correction of the torque mismatch with rpm changes is desirable for improved operation and mission effectiveness.

Engine Air Starts

45. The objective of these tests was to ensure that the T55-L-11 engines could be air started within the operational flight envelope of the helicopter, following the procedures specified in the operator's manual. Air starts were performed at approximately minimum power-required airspeed at pressure altitudes of 5,000, 10,000, and 15,000 feet. Starts were conducted on each engine with at least 3 minutes between starts to allow stabilization of the combustion chamber temperatures.

46. Either engine could be restarted at all altitudes tested. The procedure contained in the operator's manual was found to be satisfactory for all conditions tested. The engine air start characteristics of the CH-47C helicopter are satisfactory for operational use.

Engine Characteristics

47. The power-available curves used for performance tests were based on the T55-L-11 engine model specification (ref 6, app 1) and are presented in figures 46 through 57, appendix VI. A zero-degree engine inlet temperature rise and a ram pressure rise as determined from Boeing-Vertol tests were used in calculating the installed power available. The inlet characteristics curves are presented in figures 59 and 60.

48. The engine characteristics of the three test engines (LE 19110, LE 19146, and LE 19258) showed that the engines were performing to the level of the engine calibrations shown in figures 61 through 71, appendix VI. However, during the performance test program, the #2 engine showed a high vibration level which necessitated changing engines as well as exchanging locations from one side to the other. Table 7 is a list of component failures that resulted during the test program. Due to high vibration levels encountered during the test program and other incidents at other test agencies, Lycoming Division of Avco Corporation (Lycoming) is now in the process of modifying the T55-L-11 engines.

Table 7. #2 Engine Component Failure Summary (Selected Items).

Nature of Problem	Date	Component Type
Fuel temperature probe cracked	31 Oct 69	Production
Broken fire detection element	12 Nov 69	Production
Fuel temperature probe cracked	18 Nov 69	Test
Fuel temperature probe cracked	24 Nov 69	Test
Fuel temperature wire broken	28 Nov 69	Test
Broken fire detection element	3 Dec 69	Production
Fuel temperature wire broken	3 Dec 69	Test
Excessive engine vibration	29 Dec 69	Production
Broken fire detection element	30 Dec 69	Production
N_2 actuator failure	5 Jan 70	Production
N_2 actuator failure	13 Jan 70	Production
Fuel temperature wire broken	10 Apr 70	Test
Engine oil cooler assembly ruptured	30 Apr 70	Production
Igniters worn excessively	30 Apr 70	Production
Burner can brackets broken	30 Apr 70	Production
Fuel flowmeter case failed	8 Jun 70	Test
Anti-ice hot air valve threads stripped	15 Jun 70	Production
Tail pipe cracked at seam	15 Jun 70	Production
Engine forward and aft mount bearings scored	15 Jun 70	Production
Transmission cover retainer assembly worn beyond limits	15 Jun 70	Production

Torquemeter System Accuracy

49. The engine torquemeter system accuracy stated by Boeing-Vertol and by Lycoming was ±2 percent. During this evaluation, inaccuracies up to 7 percent were recorded. After the aircraft was received from the contractor and prior to the start of the performance tests, the torquemeter system was modified by Boeing-Vertol personnel to improve the accuracy. Late in the test program, reworked torquemeter power supply units were installed. The torquemeter system accuracies experienced at various times throughout the test program are shown in figures 72 through 77, appendix VI. Lycoming and Boeing-Vertol experienced similar difficulties with aircraft at the Boeing-Vertol facility which resulted in a thorough investigation of the problem. Boeing-Vertol now claims to have improved the accuracy of the system to within ±3 percent.

50. To compensate for inaccuracies, corrections were applied to observed engine torquemeter readings during all portions of the test program to ensure that the desired torque was being obtained. From an operational standpoint, torquemeter inaccuracies resulting in excessively high torquemeter readings prevent the operational pilot from achieving the maximum capability of the helicopter. Also, inadvertent high torquemeter readings would cause the pilot to believe that the torque limits as specified in the operator's manual have been exceeded. The reduced helicopter capability and the unnecessary inspections or component changes resulting from this misinformation detract from the mission effectiveness of the aircraft. The torquemeter system of the CH-47C aircraft with the T55-L-11 engines is unsatisfactory for operational use, and correction is mandatory for successful accomplishment of the intended mission.

Cruise Guide Indicator System

51. The CGI system was installed for system evaluation purposes and was used as an aid in conducting the tests.

52. The CGI system monitors the axial loads on the fixed link and pivoting actuator in the aft rotor-fixed control system. These actuators are strain gaged to measure axial loads and transmit them to the CGI indicator in the cockpit. This reading indicates the most critical fatigue load as a percent of the endurance limit of the component. A representative response of the CGI system is presented in figure 78, appendix VI. The CGI system proved to be a reliable, accurate, and repeatable indication of flight loads. The use of the CGI system allowed an increase in airspeed by allowing the pilot to fly up to a 100-percent indication, which is the limit for continuous operation. The CGI system also provided a warning to the pilot to either decrease airspeed or reduce the severity of maneuvers to minimize loads in excess of the endurance limit of the aft dynamic components. This warning is especially helpful when operating in conditions of moderate-to-heavy turbulence. Under these conditions, loads in excess of 100 percent occur quite frequently, even though the aircraft is being operated well below the envelope restrictions. The CGI system enhances the operational capability of the helicopter and should be installed in present CH-47C helicopters and incorporated in future designs.

Cabin Noise Level and Vibration

53. Throughout the test program, qualitative noise level and vibration data were obtained. In general, high vibration levels were encountered. These vibration levels became increasingly severe with increasing airspeed. Although the cabin loadings were not configured to standard vibration test loading and load distribution, the qualitative comments correlate with vibration data contained in TM 55-1520-227-20 (ref 15, app 1). The vibration levels above 120 KIAS limited crewmember effectiveness and had a fatiguing effect when sustained over a period of time (such as a cross-country flight). Correction of the objectionable cargo compartment vibrations is desirable for improved operation and mission effectiveness.

54. The excessively high cabin noise levels in the CH-47C, previously reported in USAASTA Project No. 66-28 (refs 16 and 17, app 1), required all crewmembers to wear protective helmets or sound-attenuating aural protectors. Even with this equipment, prolonged exposure to the high noise-level environment of the CH-47C produced fatigue and discomfort which decreased the effectiveness of the crew. Correction of the objectionable cabin noise levels is desired for improved operation and mission effectiveness.

Maintenance

55. Several equipment improvement recommendations (EIR's) were submitted. Not shown in EIR's is the history of problems associated with test instrumentation installed in the #1 and #2 engine compartments. These problems were the result of the exposure of test instrumentation components to a high-vibration environment and were primarily associated with the #2 engine.

56. The high frequency of repair/replacement experience with components located in the engine compartment and associated power-train area is an indication that problem areas exist which merit further investigation. High frequency of repair increases the organizational and general support maintenance required for the helicopter, decreases the mission effectiveness of the aircraft, and represents a potential inflight hazard. An engine/airframe vibration and loads survey should be conducted at the earliest possible time. The vibration characteristics of the CH-47C are unsatisfactory for operational use, and correction is mandatory.

Pitot and Static System Calibration

57. Flight tests were conducted to determine the ship's system airspeed position error and to calibrate the boom airspeed system. Based on previous test programs using the same boom system, it was found that the position error remained the same whether the aircraft was in level flight, climb, or autorotation. Tests were conducted to determine the ship's system position error. The ground speed course, trailing bomb, and pacer methods of airspeed calibration were used for level flight. The trailing bomb method was used during climbing and descending flight.

58. The pitot and static system calibration data are presented in figures 80 and 81, appendix VI. Test results show that the ship's system airspeed position error varies from a maximum of 7.5 knots at 41 knots calibrated airspeed (KCAS) to a minimum of zero knots at 110 KCAS in balanced level flight. At airspeeds below 41 KCAS, the system was unreliable. During high rates of climb (2,000 ft/min or greater), the position error increased from zero at 50 KIAS to 5 knots at 90 KIAS. Rates of climb of less than 2,000 ft/min resulted in a maximum position error of 4 knots at 46 KIAS. High rates of descent (2,000 ft/min or greater) resulted in a maximum position error of from 21.4 knots at 43 KIAS to 1 knot at 100 KIAS. Rates of descent of less than 2,000 ft/min resulted in a maximum position error of 8.5 knots at 44 KIAS and 1 knot at 100 KIAS. Although the ship's system airspeed position error fails to meet the overall requirements of MIL-I-6115A (ref 7, app I), the CH-47C system exhibits a marked improvement over the system used in the CH-47A and CH-47B and is satisfactory for operational use.

59. When cockpit vibration levels became moderately heavy, the airspeed indicator exhibited fluctuations of ±3 knots which were annoying to the pilot and made precise airspeed control more difficult (HQRS 4). The fluctuations were at a frequency of approximately 3/rev. Correction of the indicator fluctuation is desirable for improved operation and mission capability.

CONCLUSIONS

GENERAL

60. Within the scope of this test, the CH-47C helicopter performance characteristics are suitable for the transport helicopter mission, provided the inaccurate engine torquemeter system and the high engine compartment vibration levels are corrected (paras 48 and 50).

61. The CH-47C helicopter met or exceeded all contractor guarantees (para 13).

62. The pitot and static systems of the helicopter exhibit a marked improvement over the system in the CH-47A and CH-47B (para 58).

63. The improved hover performance enhances the operational capability of the CH-47C (para 17).

64. The improved dual-engine climb performance, from SL to envelope ceiling at all gross weights, enhances the capability of the CH-47C for the transport mission (para 28).

65. The CGI system enhances the operational capability of the helicopter and should be installed in present CH-47C helicopters and incorporated in future designs (para 52).

DEFICIENCIES AND SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

66. Correction of the following deficiencies is mandatory for successful accomplishment of the intended mission:

a. Torquemeter system inaccuracies of up to 7 percent above actual torque values (para 49).

b. High frequency of repair/replacement of components located in the engine compartment and associated power train area (para 56).

67. Correction of the following shortcomings is desirable for improved operation and mission capability:

a. Objectionable cockpit vibration levels which limit maximum level-flight airspeed (para 32).

b. Moderate pilot effort required to maintain optimum climb airspeeds (para 28).

c. The 3/rev airspeed indicator needle oscillations at high power settings (para 59).

d. Engine torque mismatch which results from adjusting rotor speed (para 44).

e. Use of landing gear power steering control may be lost at gross weights below 30,000 pounds (para 42).

f. Objectionable cargo compartment vibration (para 53).

g. Objectionable noise levels (para 54).

RECOMMENDATIONS

68. The data presented in this report should be used to update the operator's manual.

69. The deficiencies should be corrected on a high-priority basis.

70. The shortcomings should be corrected at the earliest possible time.

71. The following items should be included in the CH-47C operator's manual:

a. The recommended airspeed for maximum endurance should be established at 80 KIAS (para 33).

b. The climb airspeed should be increased to 80 KIAS for night operations. (para 28).

c. A discussion of the techniques used for takeoff when power available is insufficient to accomplish a vertical takeoff (para 19).

d. The level acceleration type of takeoff should not be used when power available is insufficient to permit hovering with an aft wheel height of 10 feet (para 20).

e. The technique of copilot monitoring and controlling the power parameters should be used during level-acceleration takeoffs (para 21).

72. An engine/airframe vibration and loads survey should be conducted at the earliest possible time (para 56).

73. Future tests should be conducted to define the heavy gross-weight autorotational characteristics (para 36).

APPENDIX I REFERENCES

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16. Final Report, USAASTA, Project No. 66-28, *Army Preliminary Evaluation III and IV, YCH-47C Medium Transport Helicopter*, July 1970.

17. Letter Report, Acoustical Research Branch, Engineering Research Laboratory, US Army Human Engineering Laboratories, No. 95, *Interior Noise Evaluation of the CH-47C Helicopter*, January 1969.

APPENDIX II. PHYSICAL CHARACTERISTICS OF THE CH-47C

GENERAL DIMENSIONS

Length (fuselage)	51 ft
Length (rotor blades turning)	99 ft
Height (over rotor blades at rest)	18 ft, 7.8 in.
Width of cabin	9 ft
Tread (forward gear)	10 ft, 6 in.
Tread (aft gear)	11 ft, 2 in.
Width (rotor blades turning)	60 ft

WEIGHT DATA

Empty weight (specification)	20,420 lb
Design gross weight	33,000 lb
Alternate design gross weight	46,000 lb

CENTER-OF-GRAVITY REFERENCE (figure I)

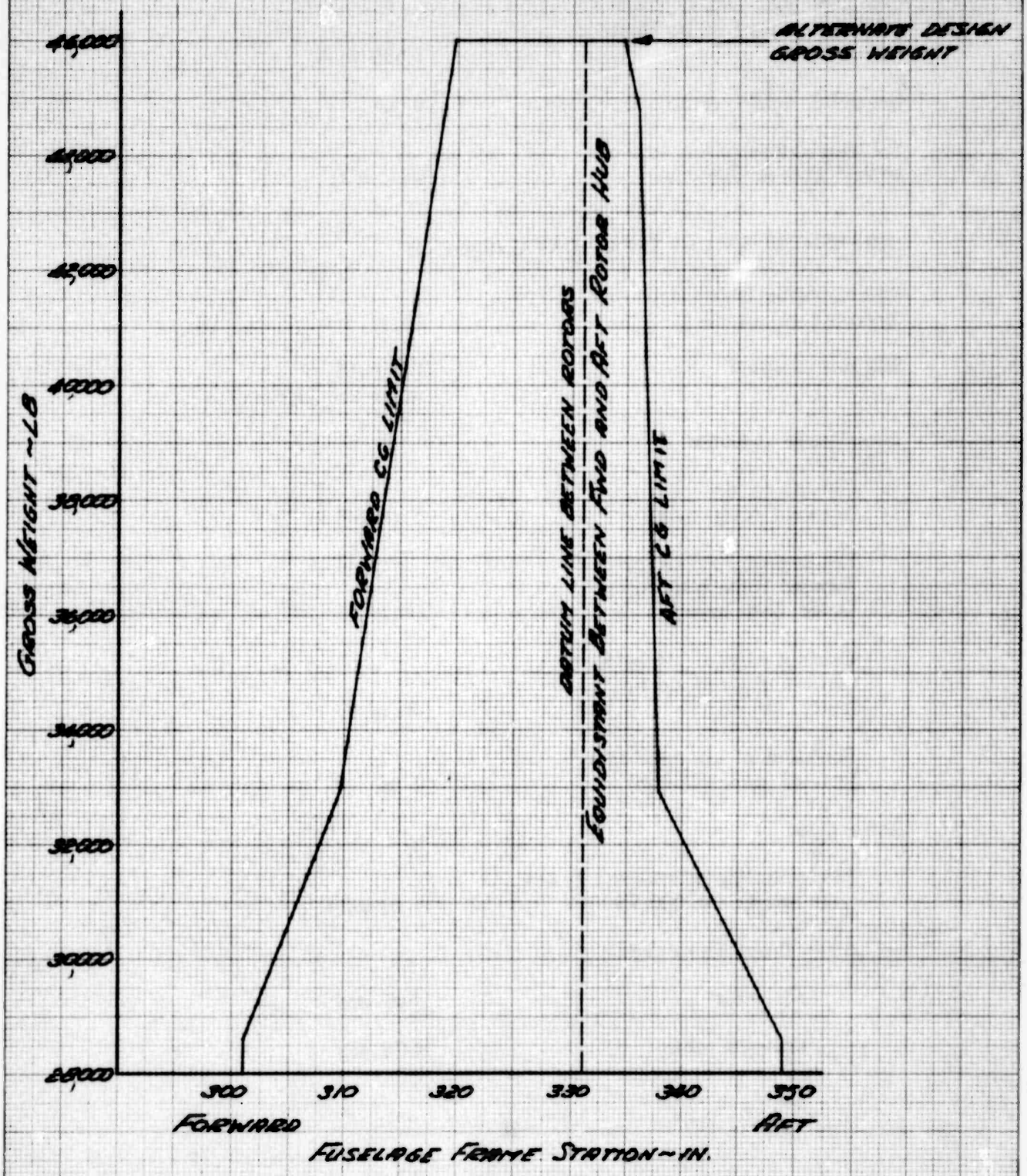
Center-of-gravity reference	FS 331.0 (centerline between rotors)
Forward limit (from cg reference)	30.0 in. forward (28,500 lb and below)
Aft limit (from cg reference)	18.0 in. aft (28,500 lb and below)

T55-L-11 ENGINE RATINGS (SL, standard day)

Maximum power	3,750 shp
Military rated power	3,400 shp
Normal rated power	3,000 shp

FIGURE I.
CENTER-OF-GRAVITY LIMITS
CH-47C
T55-L-11 ENGINES

NOTE: ENVELOPE TAKEN FROM
TM55-1520-227-10 MANUAL

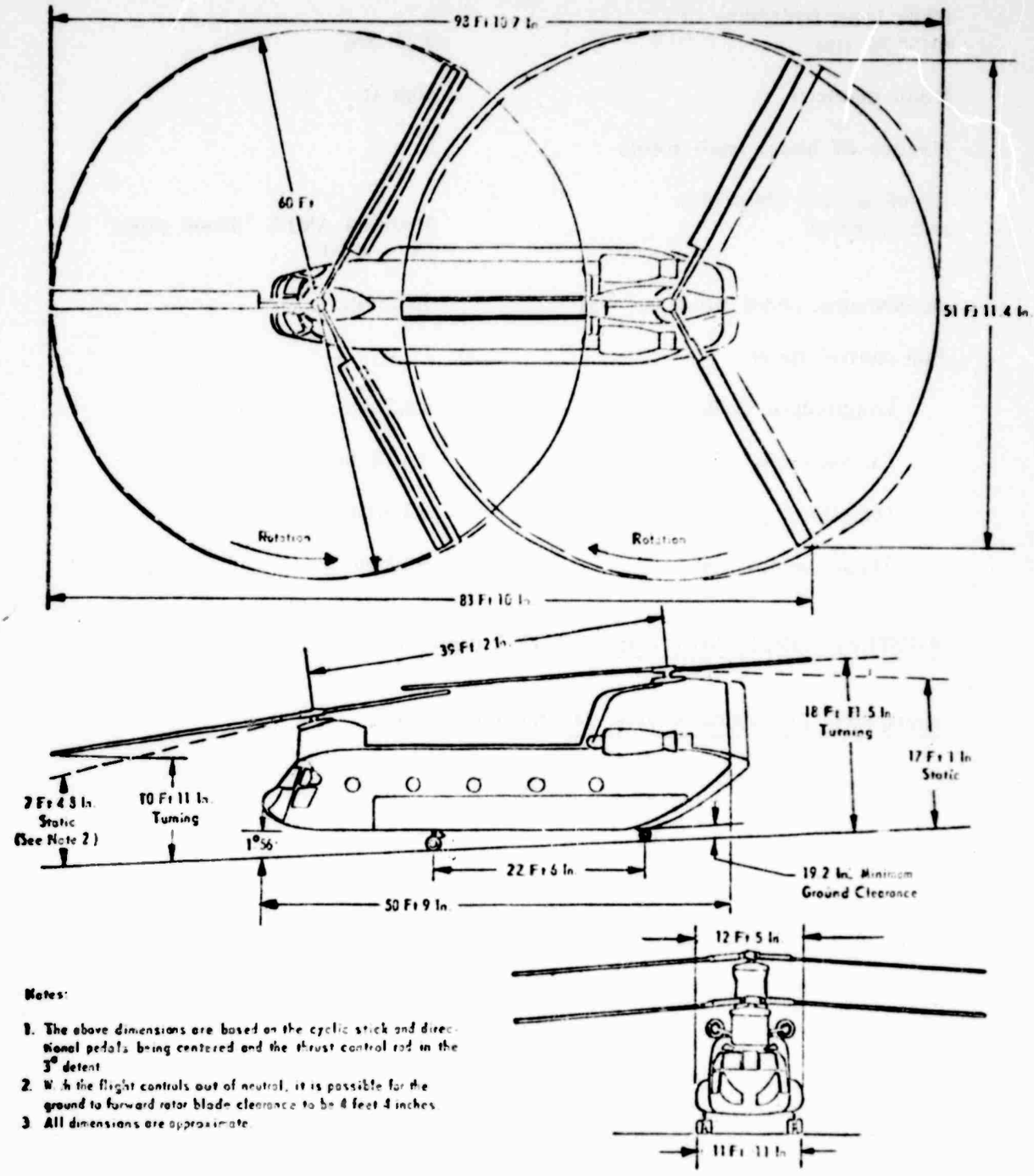


AREAS

Rotor blade area (6 at 63.1 sq ft)	379 sq ft
Swept disc area	5,000 sq ft
Geometric disc area (2 rotors at 2,827 sq ft used in performance calculations)	5,655 sq ft
Geometric solidity ratio	0.067
Sail area (cross-section area of aircraft at butt line zero)	487 sq ft

DIMENSIONS AND GENERAL DATA (figure II)

Rotor spacing (distance between centerline of rotors)	39 ft. 2 in.
Sail area centroid	FS 367.5, water line 28.6
Rotor blade clearance:	
Ground to tip (forward rotor static)	7 ft. 6.7 in.
Leading edge of aft pylon to forward rotor blade tip (rotor blade static)	16.7 in.
Leading edge of aft pylon to forward rotor blade tip (rotor turning)	40 in.
Elevation of aft rotor over forward rotor (at hub)	4 in.
Rotor data:	
Power loading at alternate design gross weight (46,000/5,920)	7.76 lb/hp
Blade droop (stop angle):	
Aft rotor	3.25 deg
Forward rotor	4.75 deg



Notes:

1. The above dimensions are based on the cyclic stick and directional pedals being centered and the thrust control rod in the 3° detent.
2. With the flight controls out of neutral, it is possible for the ground to forward rotor blade clearance to be 4 feet 4 inches.
3. All dimensions are approximate.

Figure 11 Overall dimensions

Blade coning (stop angle)	30 deg
Blade twist (centerline of rotor to tip)	9.23 deg
Rotor diameter	60.0 ft
Number of blades (each rotor)	3
Airfoil section designation and thickness	Modified AMES "droop snoot" t/c = 0.10
Aerodynamic chord (root and tip)	25.25 in.
Full control travel:	
Longitudinal cyclic	±6.5 in.
Lateral cyclic	±4.18 in.
Directional pedal	±3.60 in.
Thrust control rod	9.12 in.

AIRSPEED LIMITATION FOR 235 RPM (figure III)

AIRSPEED LIMITATION FOR 245 RPM (figure IV)

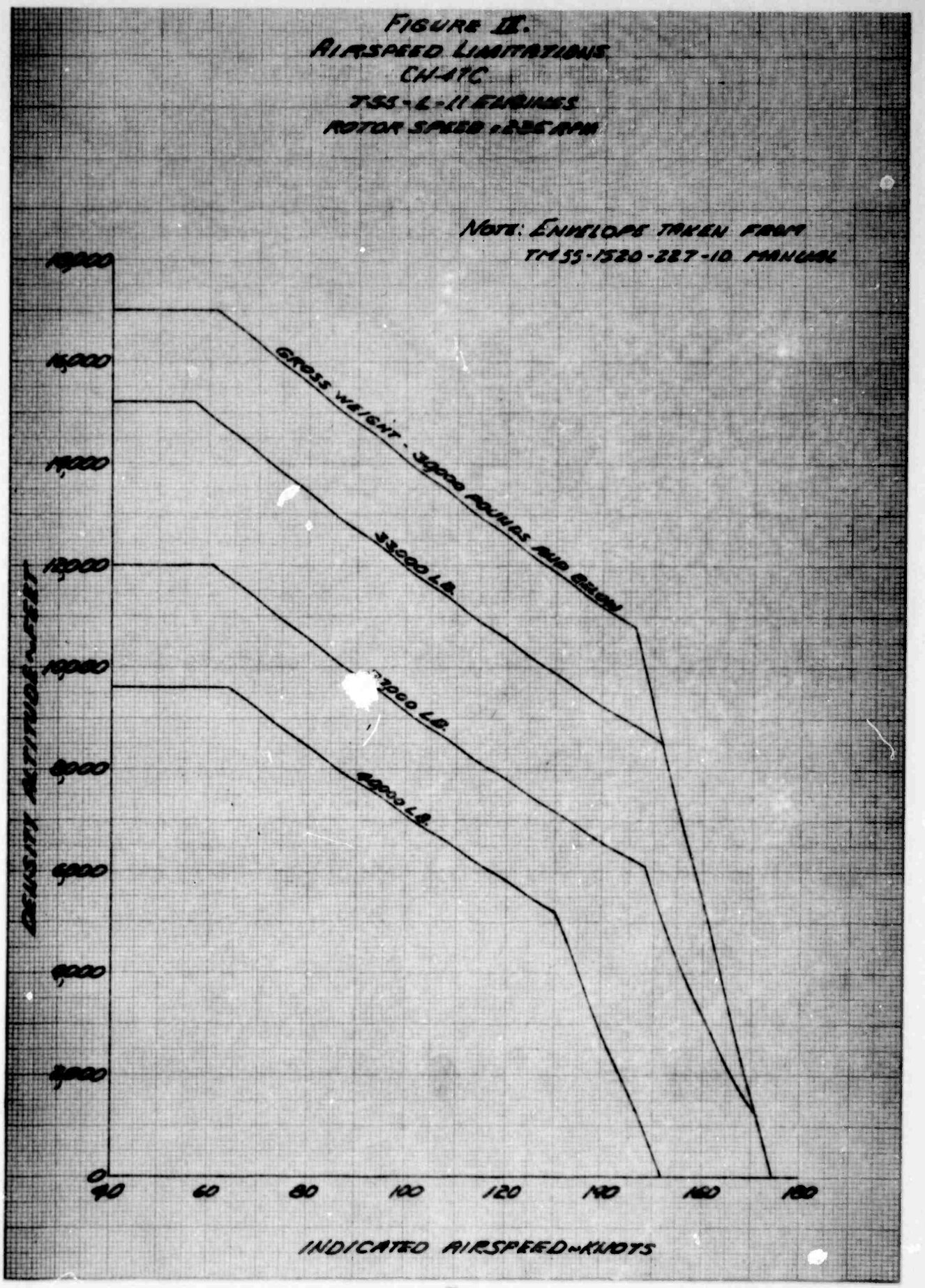


FIGURE IX.
AIRSPEED LIMITATIONS
CH-47C
T55-L-11 ENGINES
ROTOR SPEED = 235 RPM
NOTE: ENVELOPE TAKEN FROM
TM55-1520-227-10 MANUAL
GROSS WEIGHT - 30000 POUNDS AND BELOW
33000 LB.
40000 LB.
DENSITY ALTITUDE ~ FEET
18000
16000
14000
12000
10000
8000
6000
4000
2000
0
40
60
80
100
120
140
160
180
INDICATED AIRSPEED ~ KNOTS

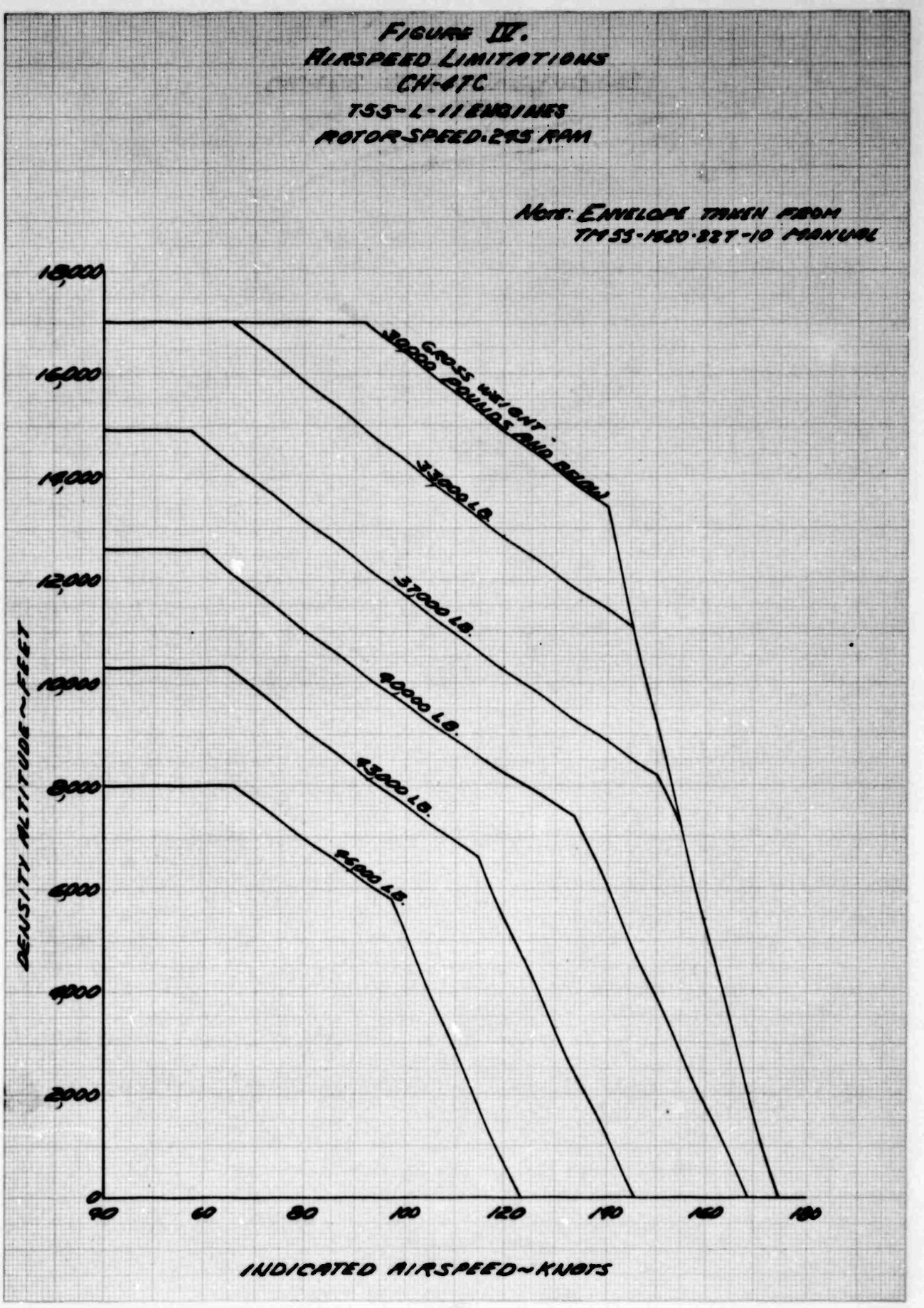


FIGURE IV.
AIRSPEED LIMITATIONS
CH-47C
T55-L-11 ENGINES
ROTOR SPEED: 225 RPM
NOTE: ENVELOPE TAKEN FROM
TM55-1520-227-10 MANUAL
GROSS WEIGHT
39,000 POUNDS AND BELOW
33,000 LB.
37,000 LB.
40,000 LB.
43,000 LB.
46,000 LB.
DENSITY ALTITUDE ~ FEET
18,000
16,000
14,000
12,000
10,000
8,000
6,000
4,000
2,000
0
40
60
80
100
120
140
160
180
INDICATED AIRSPEED ~ KNOTS

APPENDIX III. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. The equations and analysis methods used to correct test-day conditions to US standard-day conditions are briefly described in this appendix.

2. The basic nondimensional helicopter equations were used and are defined as follows:

$$C_P = \frac{SHP \times 550}{\rho A \ (\Omega R)^3} \tag{1}$$

$$C_T = \frac{W}{\rho A \ (\Omega R)^2} \tag{2}$$

$$M_{tip} = \frac{0.5925 \times \Omega R + V_T}{38.967 \times \sqrt{T}} \tag{3}$$

where: C_P = Power coefficient

SHP = Engine output shaft horsepower

ρ = Air density (slug/ft^3)

A = Total rotor geometric disc area (ft^2)

Ω = Rotor angular velocity (rad/sec)

R = Rotor radius (ft)

C_T = Thrust coefficient

W = Gross weight (lb)

μ = Advance ratio

V_T = True airspeed (kt)

M_{tip} = Advancing tip Mach number

T = Ambient temperature (°K)

3. Significant compressibility effects are encountered at high M_{tip}. In order to best account for the effects of compressibility, the average tip Mach number, given by equation 4, should be held constant.

$$\left(M_{tip}\right)_{avg} = K_1 \times R \times \frac{N_R}{\sqrt{\theta}} \tag{4}$$

where: $K_1 = \frac{0.5925}{38.967} \times \frac{2\pi}{60} \times \frac{1}{16.9706} = 9.3826 \times 10^{-5}$

Therefore, equation 1 was redefined by noting that:

$$\rho = \rho \frac{\rho_o}{\rho_o} = \sigma \rho_o = \frac{\delta}{\sqrt{\theta}} \rho_o \frac{\sqrt{\theta}}{\theta} = \rho_o \frac{\delta\sqrt{\theta}}{\sqrt{\theta}^3} \tag{5}$$

and: $$\Omega R = \frac{2\pi}{60} \times N_R \times R \tag{6}$$

where: ρ_o = SL, standard-day air density (slug/ft^3)

σ = Density ratio

δ = Pressure ratio

θ = Temperature ratio

N_R = Rotor rotational speed (rpm)

Therefore, equation 1 becomes:

$$C_P = \frac{SHP}{\delta\sqrt{\theta}} \times \frac{550}{\rho_o AR^3} \times \frac{1}{\left(\frac{2\pi}{60} \times \frac{N_R}{\sqrt{\theta}}\right)^3} \tag{7}$$

4. Using basic equations 2 and 3 and the above procedure:

$$C_T = \frac{W}{\delta} \times \frac{1}{\rho_o AR^2} \times \frac{1}{\left(\frac{2\pi}{60} \times \frac{N_R}{\sqrt{\theta}}\right)^2} \tag{8}$$

and: $$M_{tip} = K_1 \times R \times \frac{N_R}{\sqrt{\theta}} + K_2 \frac{V_T}{\sqrt{\theta}} \quad (9)$$

where: $$K_2 = \frac{1}{38.967} \times \frac{1}{16.9706} = 1.5122 \times 10^{-3}$$

Power Determination

5. The method of determining engine output shaft horsepower from calibrated engine torquemeters was not used for this program. Previous tests on CH-47 helicopters have been plagued by torquemeter inconsistency and inaccuracy. For these tests, actual output shaft horsepower for the T55 engine was determined using measured fuel flow.

6. The fuel flow was recorded on an oscillograph from which the calculated flow rate was then referred to engine inlet conditions.

$$W_f = \frac{R \times (\text{fuel spec}) \times 3600}{K_1} = \text{lb/hr} \quad (10)$$

and: $$\left(W_f\right)_{\text{referred}} = \frac{W_f}{\delta \sqrt{\theta_i}} \quad (11)$$

where: R = Number of blips per second from oscillograph

K_1 = Constant for converting blips to gallons

δ = Inlet total pressure ratio

$\sqrt{\theta_i}$ = Inlet total temperature ratio

7. From the Lycoming test stand engine calibration curve, referred shaft horsepower was found at the corresponding fuel flow. The actual shaft horsepower was determined by unreferring the referred shaft horsepower and applying corrections for ram effect and nonoptimum power turbine speed.

8. During the program, rotor torque was also recorded. Shaft horsepower was calculated as follows:

$$RHP = RT \times N_R \times K_1 \tag{12}$$

where: RT = Rotor torque from calibration (in.-lb)

N_R = Rotor speed (rpm)

K_1 = Constant = $\frac{2\pi}{60 \times 12 \times 550}$

Assumed constant transmission and accessory loss = 180 hp

Therefore:

$$SHP = RHP_{fwd\ rotor} + RHP_{aft\ rotor} + 180 \tag{13}$$

9. Comparison between the fuel flow and rotor torque calculated shaft horsepower revealed that inconsistencies existed during hover performance tests. The inconsistent power comparison was not large and, therefore, could not be detected during the level-flight performance test.

10. To reconcile these inconsistencies between fuel flow and rotor torque measured powers, the test engines were returned to Lycoming for recalibration to determine if a shift had occurred (which could account for the deviation between fuel flow and rotor torque calculated powers). Premature disassembly of one of the test engines prior to reaching Lycoming prevented its recalibration. However, the recalibration of the other engine did reveal a 2-percent shift which could modify the fuel-flow power so as to provide better correlation between the two data sources. Since the time period of the shift could not be specified, all performance data were determined from fuel flow and converted to standard-day conditions using the engine model specification.

HOVER

11. Hover performance was determined by using $N_R/\sqrt{\theta}$ as the test variable at each gross weight. The tethered hover technique was used, and limited free-flight hover data were gathered to substantiate the tethered hover technique of data gathering.

12. C_P versus C_T at OGE hover was plotted for constant M_{tip} number. As the M_{tip} increased, compressibility effects were noted. Using OGE hover data, lines

of constant M_{tip} were drawn. These fairings were used to construct figure 12, appendix VI, with $C_{P_{compressible}} - C_{P_{incompressible}}$ (ΔC_P) as a function of M_{tip} and thrust coefficient. Compressibility effects were observed to begin at an M_{tip} of 0.563, or $N_R/\sqrt{\theta}$ of 200 rpm, and increase as C_T and M_{tip} increased.

13. C_P versus C_T at a 5-foot hover showed no compressibility effects, regardless of M_{tip} or C_T. As the hover height increased, compressibility effects became more pronounced.

14. Figure AA was derived from the assumption that as wheel height increased, an increasing percentage of compressibility power increment should be applied to the incompressible power. A logarithmic variation was made based on zero percent at 5 feet and 100 percent at 80 feet (approximately OGE). This percentage was applied to the correction obtained from figure 12, appendix VI, at the appropriate wheel heights for presenting incompressible hover data.

15. Comparison of the SL hover data with the data from a high altitude (9500 feet) revealed that some inconsistencies existed. To resolve the discrepancies, a review of all previous hover data gathered on the CH-47B (same rotor system) was conducted, and the test engines were returned to the manufacturer for recalibration (para 10).

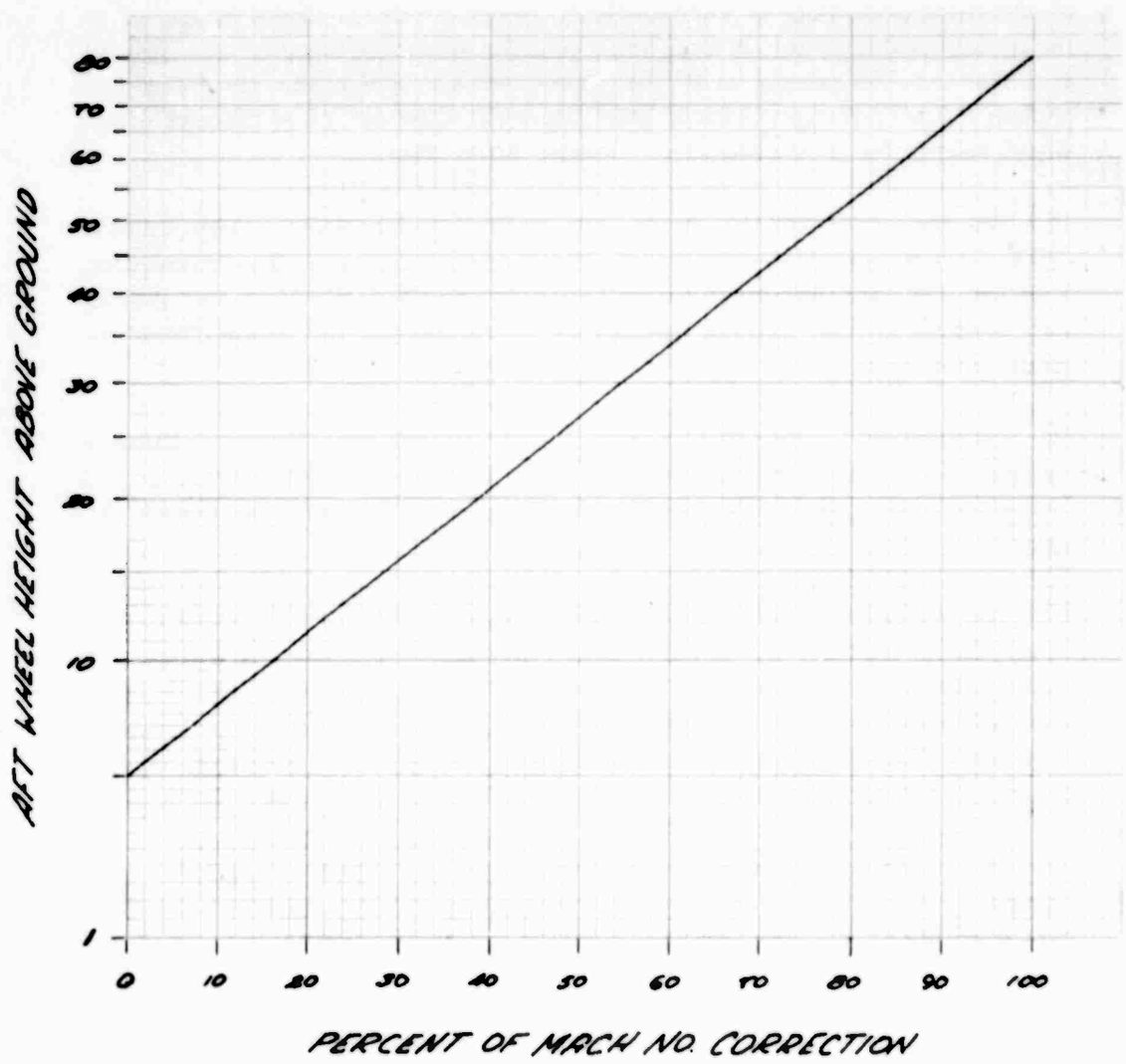
16. Examination of the CH-47B (ref 17, app I) hover data did not exclusively validate either the CH-47C A&FC SL or high-altitude data. However, its "incompressible" performance level coincides more closely with the CH-47C SL data.

17. The anomalies of the hover data have required that judgment-biased fairings be applied to all the CH-47C hover data. These results are presented in figures 1 through 11, appendix VI. It should be noted that either this approach or an approach based on statistical fairings of all CH-47B/C data will indicate performance in excess of the contractor's guarantees. Further testing and/or analysis is required to resolve measured discrepencies in the CH-47 hover performance.

18. The summary hovering performance (figs. 1 through 4, app VI) was calculated using the nondimensional hovering curves and the power-available curve shown in figures 48 and 51.

FIGURE AA.

PERCENT OF MACH NO. CORRECTION
VS
AFT WHEEL HEIGHT ABOVE GROUND
CH-47C USA S/N 68-15859



Takeoff

19. Takeoff tests were conducted to determine takeoff performance using the level acceleration technique. Takeoffs were initiated from a hover using a 10-foot wheel height as a reference for power required. Maximum power was applied during the acceleration from hover and maintained until a 100-foot obstacle was cleared. Rotation to climb attitude was initiated approximately 5 knots below climbout airspeed.

20. A series of takeoffs were flown to provide a range of ΔCp where ΔCp is defined as the difference between the test maximum power available (Cp_{aval}) at test ambient conditions and the power required (Cp_{rqrd}) to hover at 10 feet ($\Delta Cp = Cp_{aval} - Cp_{rqrd}$). Gross weight was varied depending on atmospheric conditions to obtain a range of ΔCp values. A Fairchild Flight Analyzer camera was used to obtain a graphical time history of each takeoff. True ground speed, height, and horizontal distance were measured from the time histories, and the true climbout airspeed was derived.

21. For each ΔCp, a plot of horizontal distance to clear a 100-foot obstacle versus true climbout airspeed was constructed. The plots for the various ΔCp's were made into a carpet plot which relates the takeoff performance of the aircraft. The carpet plot can then be used to determine the best climbout airspeed with a corresponding distance required to clear a 100-foot obstacle using the level acceleration technique. Also, takeoffs were constructed as a function of thrust coefficient as related to ΔCp.

Climbs

22. All climbs were performed at the best climb airspeed which was determined from level-flight performance data. Best climb airspeed was assumed to be the airspeed for minimum power required in level flight.

23. Sawtooth climbs were flown to determine the power correction coefficient (K_p) and weight correction coefficient (K_w). In climbs to service ceiling, K_p and K_w were used to determine the corrections to rate of climb caused by the differences in shaft horsepower and in gross weight, respectively, between test and standard conditions. These differences occur when the power and fuel consumption of an installed test engine for test-day conditions are corrected to an engine model specification for standard-day conditions.

24. During the data reduction phase of the program, it was found that for light gross weight and high rates of climb (greater than 1100 ft/min), K_p appeared not to be a constant. The power correction for climbs (fig. BB) was obtained from figure 24, appendix VI, for rates of climb below 1400 ft/min. The dashed portion of the curve in figure BB was derived from the tapeline rate-of-climb data obtained from the lightweight climb. It was assumed that:

$$R/C_t \cong R/C_{max} \tag{14}$$

where: $R/C_{max} = K_p \times \frac{\Delta SHP}{W_s} \times 33{,}000$

and: $\Delta SHP = SHP_{aval} - SHP_{rqrd \text{ for level flight}}$

This assumption was based on the fact that R/C_t is much greater than either $\Delta\ R/C_p$ or $\Delta\ R/C_w$ in the equation:

$$R/C_s = R/C_t + \Delta\ R/C_p + \Delta\ R/C_w \tag{15}$$

Therefore:

$$R/C_s \cong R/C_t$$

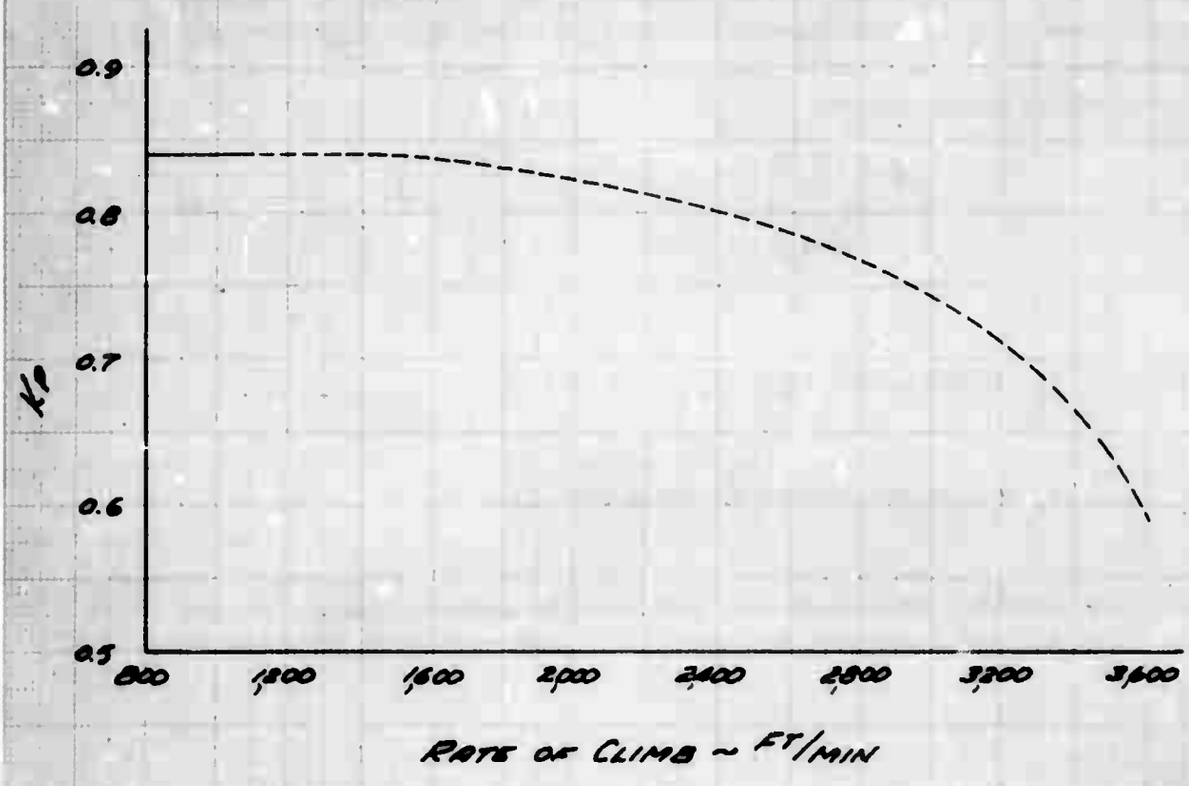
where: $R/C_s = R/C_{max}$

Then: $$R/C_t \cong K_p \times \frac{\Delta SHP}{W_s} \times 33{,}000 \tag{16}$$

This equation can be solved for K_p. Figure BB shows that K_p is a function of rates of climb and cannot be verified by the limited sawtooth climb data at light weights. Future tests should be conducted at light weights (26,000 lb) and high rates of climb (2000 ft/min or greater) to establish maximum climb capability.

FIGURE BB.
POWER CORRECTION FOR CLIMBS
CH-47C USA S/N 68-15839
MID CG

$$K_p = \frac{(\Delta R/C)(\text{GROSS WEIGHT})}{(\Delta \text{SHP})(33{,}000)}$$



25. The equations used for power and weight corrections are as follows:

$$\Delta R/C_p = K_p \times \frac{\Delta SHP}{W} \times 33{,}000 \qquad (17)$$

$$\Delta R/C_w = K_w \times SHP_s \times 33{,}000 \left(\frac{1}{W_s} - \frac{1}{W_t}\right) \qquad (18)$$

where: ΔSHP = Standard-day shaft horsepower available as defined in the engine model specification minus test measured shaft horsepower

W_t = Test gross weight

SHP_s = Standard-day shaft horsepower available as defined in the engine model specification

W_s = Standard gross weight

26. Continuous climbs were conducted to service ceilings or to a 15,000-foot pressure altitude (Hp), whichever was reached first. The 15,000-foot Hp limitation was imposed because of the possibility of the flight control hydraulic boost pump cavitating. The indicated rate of climb (dHp/dt) was corrected to tapeline rate of climb (R/C_t) by the equation:

$$R/C_t = \left(\frac{dH_p}{dt}\right)\left(\frac{T_{a_t}}{T_{a_s}}\right) \qquad (19)$$

where: T_{a_t} = Test ambient air temperature (°K)

T_{a_s} = Standard ambient air temperature (°K)

27. The standard rate of climb was determined by correcting the tapeline rate of climb for shaft horsepower and gross weight differences using equations 17 and 18.

Therefore:

$$R/C_s = R/C_t + \Delta R/C_p + \Delta R/C_w \tag{20}$$

LEVEL FLIGHT

28. Level flight speed-power performance was determined using equations 7 and 9. Each speed-power polar was flown maintaining a constant referred gross weight (W/δ) and referred rotor speed ($N_R/\sqrt{\theta}$). A constant W/δ was maintained by decreasing ambient pressure ratio (δ) as the aircraft gross weight decreased due to fuel burnoff. Rotor speed was also varied to maintain a constant $N_R/\sqrt{\theta}$ as the outside air temperature varied.

29. The raw data were reduced to referred terms: $SHP_t/\delta\sqrt{\theta}$, $V_T/\sqrt{\theta}$, W/δ, and $N_R/\sqrt{\theta}$. Each point was then corrected to unaccelerated flight, zero rate of climb, aim W/δ, aim $N_R/\sqrt{\theta}$, and equivalent flat plate area due to nonproduction aircraft configuration. These corrections are defined as follows:

a. Acceleration-Deceleration Correction.

$$F = ma \tag{21}$$

where: F = Force

m = Mass (W/g)

a = Acceleration ($\Delta V_T/\Delta t$)

$$\Delta F = \frac{W}{g} \times \frac{\Delta V_T}{\Delta t} \times 1.6889 \tag{22}$$

where: $g = 32.174$ ft/sec^2

$$\Delta F \times V_T = \frac{W}{g} \times \frac{\Delta V_T}{\Delta t} \times V_T \times 1.6889^2 \tag{23}$$

$$\Delta SHP = \frac{W}{g} \times \frac{\Delta V_T}{\Delta t} \times V_T \times K \tag{24}$$

where: K = Constant to convert units to shaft horsepower

$$\frac{\Delta SHP}{\delta\sqrt{\theta}} = \frac{1}{\delta\sqrt{\theta}} \times \frac{\sqrt{\theta}}{\sqrt{\theta}} \times W \times \frac{\Delta V_T}{\Delta t} \times V_T \times K_1 \qquad (25)$$

where: $K_1 = \frac{1.6889^2}{32.174 \times 33{,}000} = 2.6865 \times 10^{-6}$

$$\frac{\Delta SHP}{\delta\sqrt{\theta}} = \frac{W}{\delta} \times \frac{V_T}{\sqrt{\theta}} \times \frac{V_T}{\sqrt{\theta}} \times \frac{\sqrt{\theta}}{\Delta t} \times K_1 \qquad (26)$$

or:

$$\frac{\Delta SHP}{\delta\sqrt{\theta}} = \frac{W}{\delta} \times \frac{\Delta V_T/\sqrt{\theta}}{\Delta t} \times \frac{V_T}{\sqrt{\theta}} \times \sqrt{\theta} \times K_1 \qquad (27)$$

where: $\frac{\Delta SHP}{\delta\sqrt{\theta}}$ = Referred shaft horsepower correction (shp)

$\frac{W}{\delta}$ = Referred test gross weight (lb)

$\frac{\Delta V_T/\sqrt{\theta}}{\Delta t}$ = Change in referred true airspeed per unit change of time (kt/sec)

$\frac{V_T}{\sqrt{\theta}}$ = Referred true airspeed (kt/sec)

A plot of $V_T/\sqrt{\theta}$ versus time was constructed, and a line was faired through the points. At a selected $V_T/\sqrt{\theta}$, the slope $\Delta V_T/\sqrt{\theta} \div \Delta t$ was determined. By using the value of $\Delta V_T/\sqrt{\theta} \div \Delta t$ and the selected $V_T/\sqrt{\theta}$ in equation 27, the difference in $SHP/\delta\sqrt{\theta}$ was computed for unaccelerated flight.

b. Rate-of-Climb or Rate-of-Descent Correction.

From equation 17:

$$\Delta R/C_p = K_p \times \frac{\Delta SHP}{W_t} \times 33{,}000 \tag{28}$$

$$\Delta SHP = \frac{\Delta R/C_p \times W_t}{K_p \times 33{,}000} \tag{29}$$

$$\frac{\Delta SHP}{\delta\sqrt{\theta}} = \frac{1}{\delta\sqrt{\theta}} \times \frac{\Delta R/C \times W_t}{K_p \times 33{,}000} \tag{30}$$

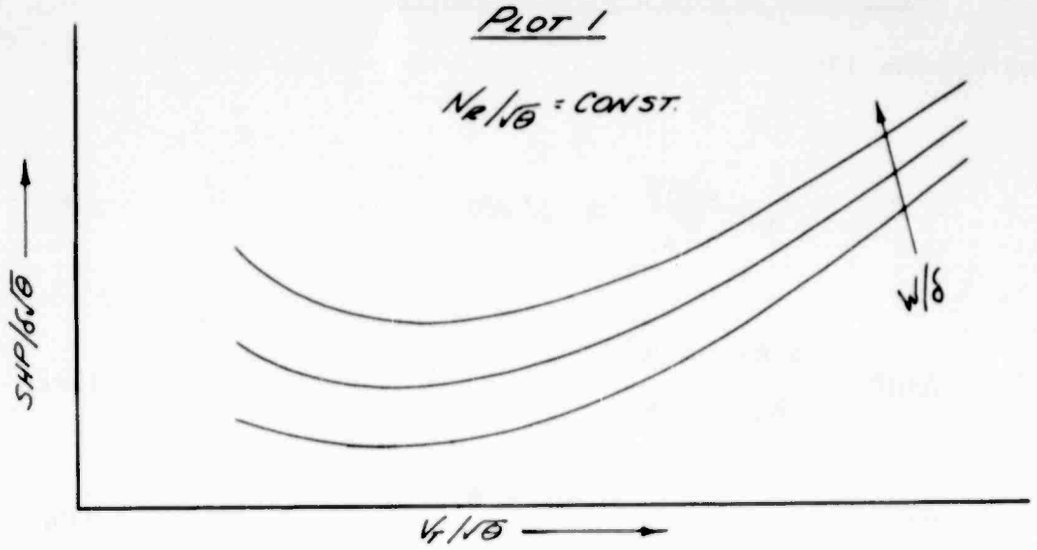
$$\frac{\Delta SHP}{\delta\sqrt{\theta}} = \frac{\frac{\Delta R/C}{\sqrt{\theta}} \times \frac{W_t}{\delta}}{K_p \times 33{,}000} \tag{31}$$

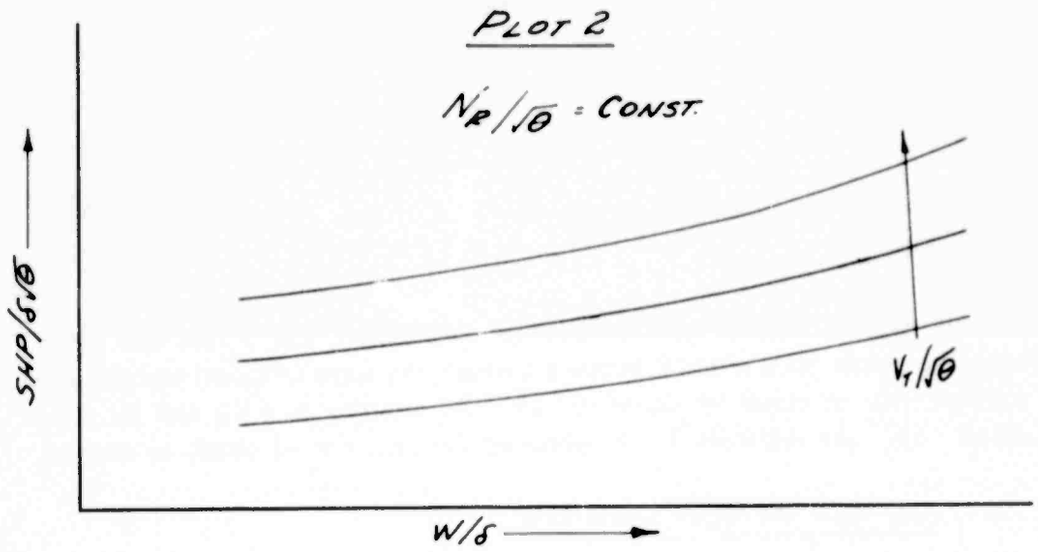
A plot of pressure altitude versus time was constructed, and a line was faired through the points. At a selected pressure altitude, the slope (dH_p/dt) was changed to tapeline rate of climb by equation 19. By referring $\Delta R/C_t$ and by using equation 31, the $\Delta SHP/\delta\sqrt{\theta}$ was obtained for zero rate of climb or descent.

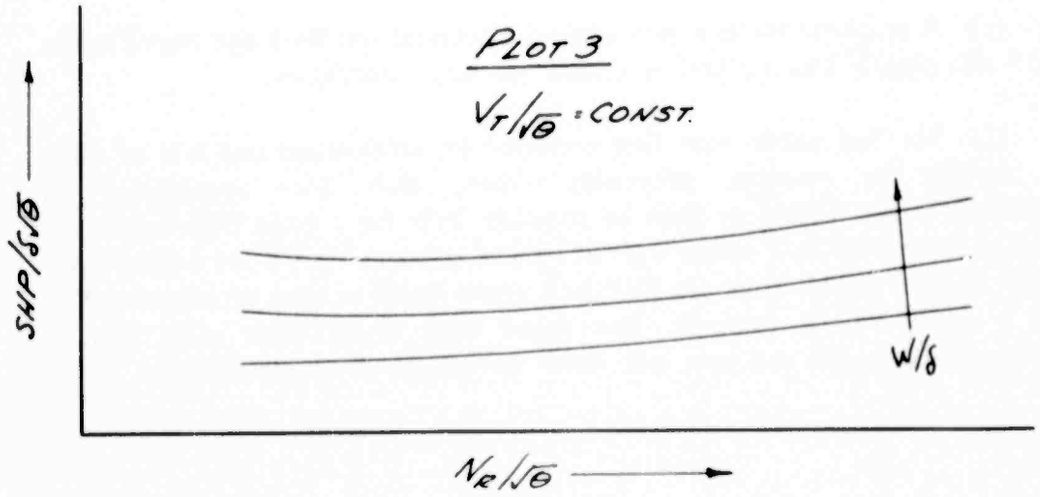
c. Aim W_t/δ and $N_R/\sqrt{\theta}$ Correction.

(1) A graphical solution was applied to correct test W_t/δ and $N_R/\sqrt{\theta}$ to aim W_t/δ and $N_R/\sqrt{\theta}$ This method is invalid for large corrections.

(2) The test points were first corrected for acceleration and rate of climb or descent as described previously. Then, plots were constructed for $SHP/\delta\sqrt{\theta}$ versus $V_T/\sqrt{\theta}$ at lines of constant W_t/δ for a given $N_R/\sqrt{\theta}$ (plot 1); next, plots for $SHP/\delta\sqrt{\theta}$ versus W_t/δ at lines of constant $V_T/\sqrt{\theta}$ for a given $N/\sqrt{\theta}$ (plot 2); and finally, plots for $SHP/\delta\sqrt{\theta}$ versus $N_R/\sqrt{\theta}$ at lines of constant W_t/δ for a given $V_T/\sqrt{\theta}$ (plot 3). The faired lines of all three plots may be cross-referenced. The last plot will show the effects of compressibility.


PLOT 1
$N_R/\sqrt{\theta}$ = CONST.
W/δ
$SHP/\delta\sqrt{\theta}$
$V_T/\sqrt{\theta}$


PLOT 2
$N_R/\sqrt{\theta}$ = CONST.
$V_T/\sqrt{\theta}$
$SHP/\delta\sqrt{\theta}$
W/δ


PLOT 3
$V_T/\sqrt{\theta}$ = CONST.
W/δ
$SHP/\delta\sqrt{\theta}$
$N_R/\sqrt{\theta}$

(3) At the aim W/δ, enter plot 1 and find the slope ($\Delta SHP/\delta\sqrt{\theta} \div \Delta W/\delta$) at each $V_T/\sqrt{\theta}$. Construct a plot of $\Delta SHP/\delta\sqrt{\theta} \div \Delta W/\delta$ versus $V_T/\sqrt{\theta}$. At the test $V_T/\sqrt{\theta}$, find the corresponding $\Delta SHP/\delta\sqrt{\theta} \div \Delta W/\delta$, which, in turn, is multiplied by the difference of test to aim W/δ. The resultant $\Delta SHP/\delta\sqrt{\theta}$ is the W/δ correction.

(4) The same procedure is used to solve for the $\Delta SHP/\delta\sqrt{\theta}$ for a $\Delta N_R/\sqrt{\theta}$. Plot 3 is used, and a plot of $\Delta SHP/\delta\sqrt{\theta} \div \Delta N_R/\sqrt{\theta}$ versus $V_T/\sqrt{\theta}$ is constructed.

d. Equivalent Flat Plate Area Correction.

(1) The incremental power required due to an aircraft configuration change is calculated using the following equation:

$$\frac{\Delta SHP}{\delta\sqrt{\theta}} = K \times F_e \times \left(\frac{V_T}{\sqrt{\theta}}\right)^3 \qquad (32)$$

where: $F_e = C_D \times A$ for the nonstandard equipment as determined experimentally (ft^2)

and: C_D = Coefficient of drag

A = Area

$\frac{V_T}{\sqrt{\theta}}$ = Referred true airspeed (kt)

K = Constant to convert units to shaft horsepower

(2) During the CH-47C A&FC test program, an equivalent flat plate area of 2.1 square feet was used to correct for the rotor torque strain gages on two of the rotor heads and the test boom system used for test airspeed and altitude instruments. This figure was based on Boeing-Vertol's calculations and modified for the smaller boom system used during this test program.

APPENDIX IV. TEST INSTRUMENTATION

The following instrumentation were installed in the test helicopter:

PILOT PANEL

Boom airspeed
Sensitive rotor speed
Sensitive boom altimeter
Rate-of-climb indicator
Cruise guide indicator
Photopanel event switch
Record light

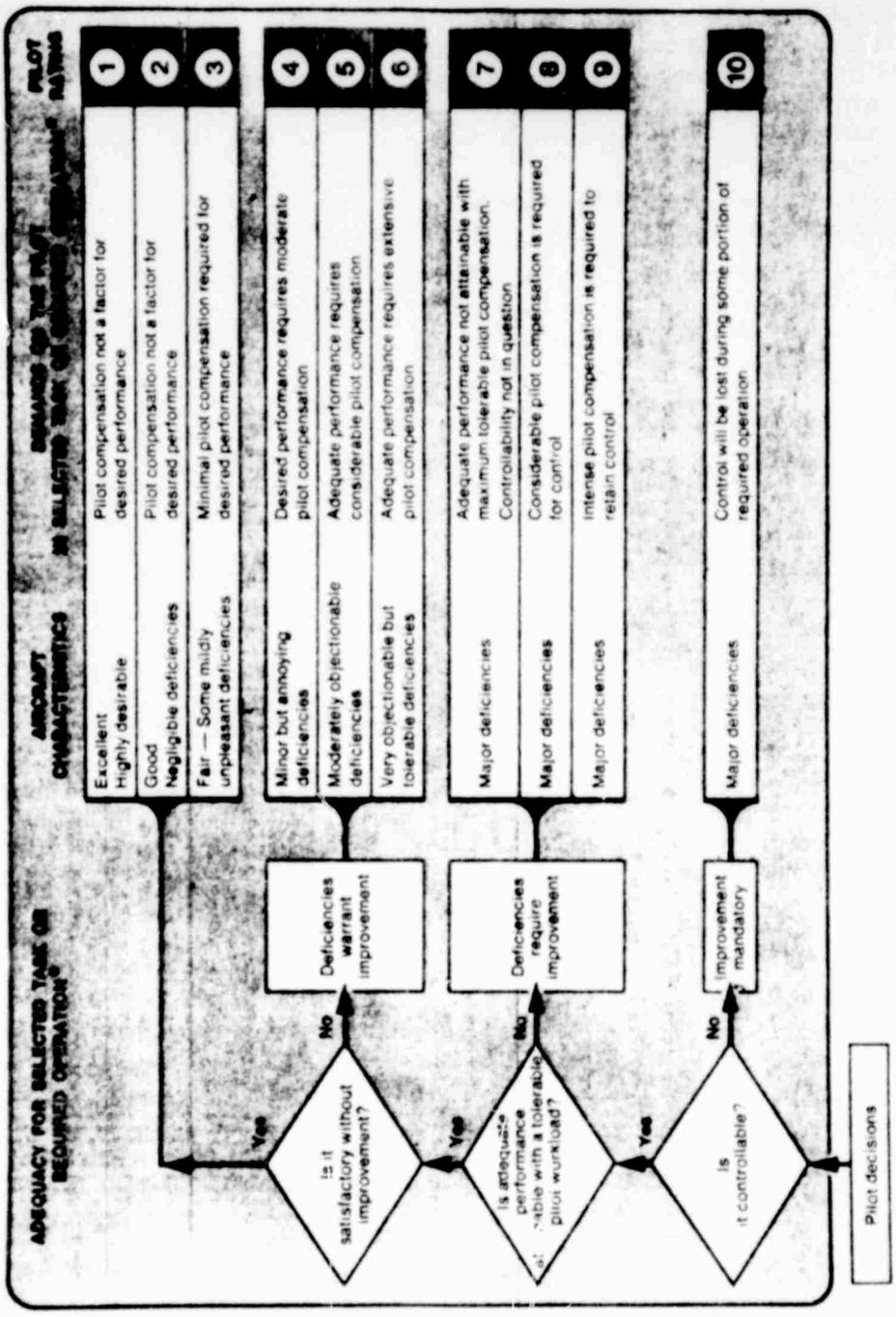
PHOTOPANEL

Boom airspeed
Ship's system airspeed
Boom altimeter
Ship's system altimeter
Sensitive rotor speed
Gas producer speed, N_1 (both engines)
Compressor inlet temperature
Free air temperature
Rate-of-climb indicator
Fuel-flow stepper motor (both engines)
Fuel totalizer (both engines)
Power turbine inlet temperature (both engines)
Torque (both engines)
Fuel temperature (both engines)
Load cell indicator
Time of day
Hayden timer
Correlation counter
Camera counter
Oscillograph counter
Event light

OSCILLOGRAPH

Rotor speed (blip)
Engine fuel flow (cycles) (both engines)
Rotor torque (both rotors)
Pilot event
Engineer event
Gas producer speed, N_1 (both engines)
Cruise guide indicator
Aft pivoting link actuator
Aft fixed-link actuator
Inlet guide vane
Gas producer arm (both engines)
Camera blip

APPENDIX V. HANDLING QUALITIES RATING SCALE



APPENDIX VI. TEST DATA

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FIGURE 1
10-FOOT HOVER CAPABILITY COMPARISON
CH-47C USA S/N 68-15859
TWO T55-L-11 ENGINES
MAXIMUM RATED POWER
ROTOR SPEED = 245 RPM

NOTES:
1. CURVE BASED ON FIGURE 6 AND 51
2. FIGURE 12 USED TO ADJUST FOR COMPRESSIBILITY EFFECTS.
3. TRANSMISSION LIMITED POWER BASED ON 6050 SHP.
4. HOVER HEIGHT = 10 FT FROM GROUND LEVEL TO BOTTOM OF AFT REAR WHEEL.

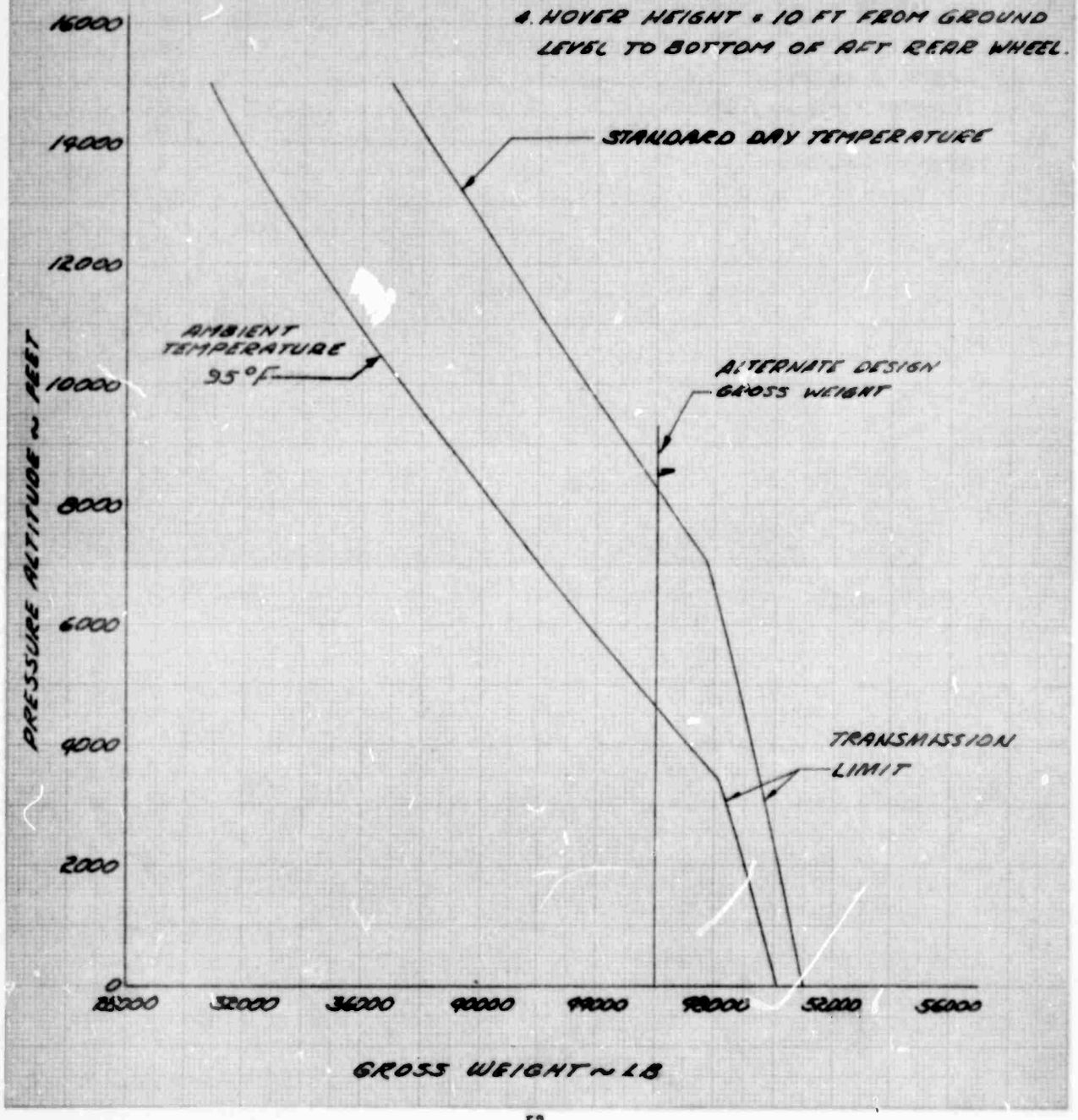


FIGURE 2
OGE HOVER CAPABILITY COMPARISON
CH-47C USA S/N 68-15859
TWO T55-L-11 ENGINES
MAXIMUM RATED POWER
ROTOR SPEED: 235 RPM

NOTES:
1. CURVE BASED ON FIGURE 6 AND 48
2. FIGURE 12 USED TO ADJUST FOR COMPRESSIBILITY EFFECTS.
3. TRANSMISSION LIMITED POWER BASED ON 5800 SHP.

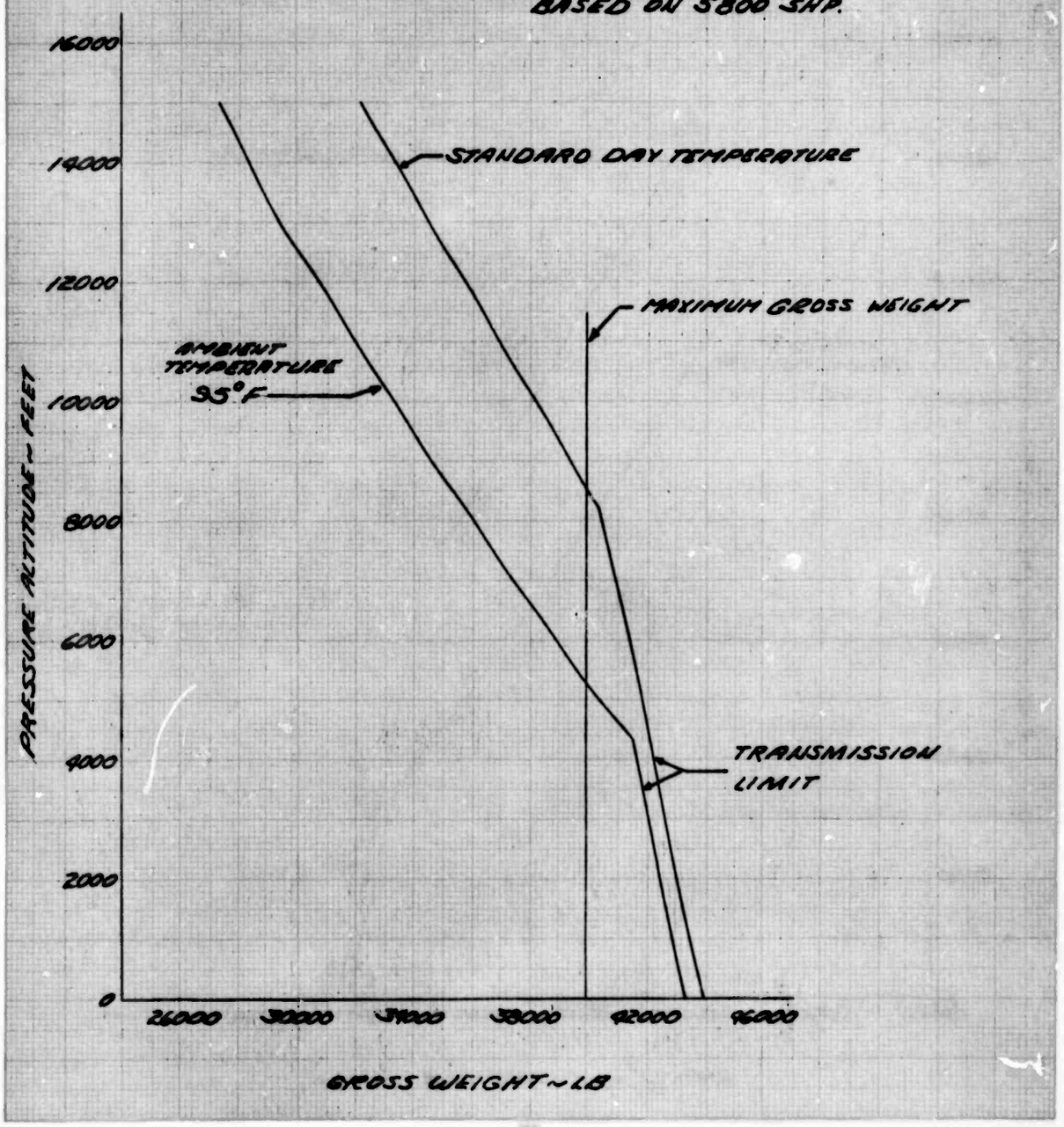


FIGURE 3
OGE HOVER CAPABILITY COMPARISON
CH-47C USA S/N 68-15859
TWO T55-L-11 ENGINES
MAXIMUM RATED POWER
ROTOR SPEED = 245 RPM

NOTES:
1. CURVE BASED ON FIGURE 6 AND 51
2. FIGURE 12 USED TO ADJUST FOR COMPRESSIBILITY EFFECTS.
3. TRANSMISSION LIMITED POWER BASED ON 6050 SHP.

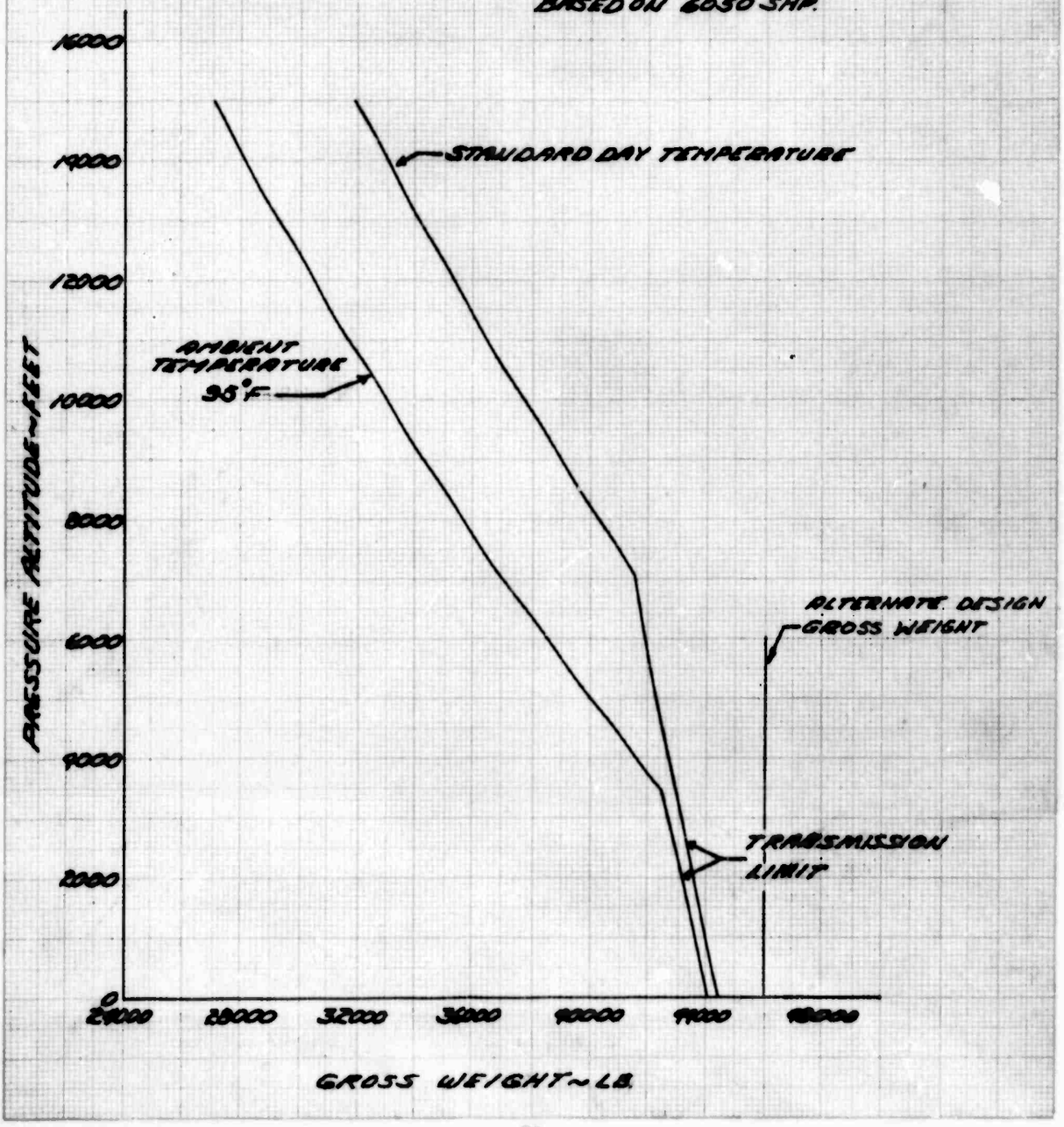


FIGURE 4

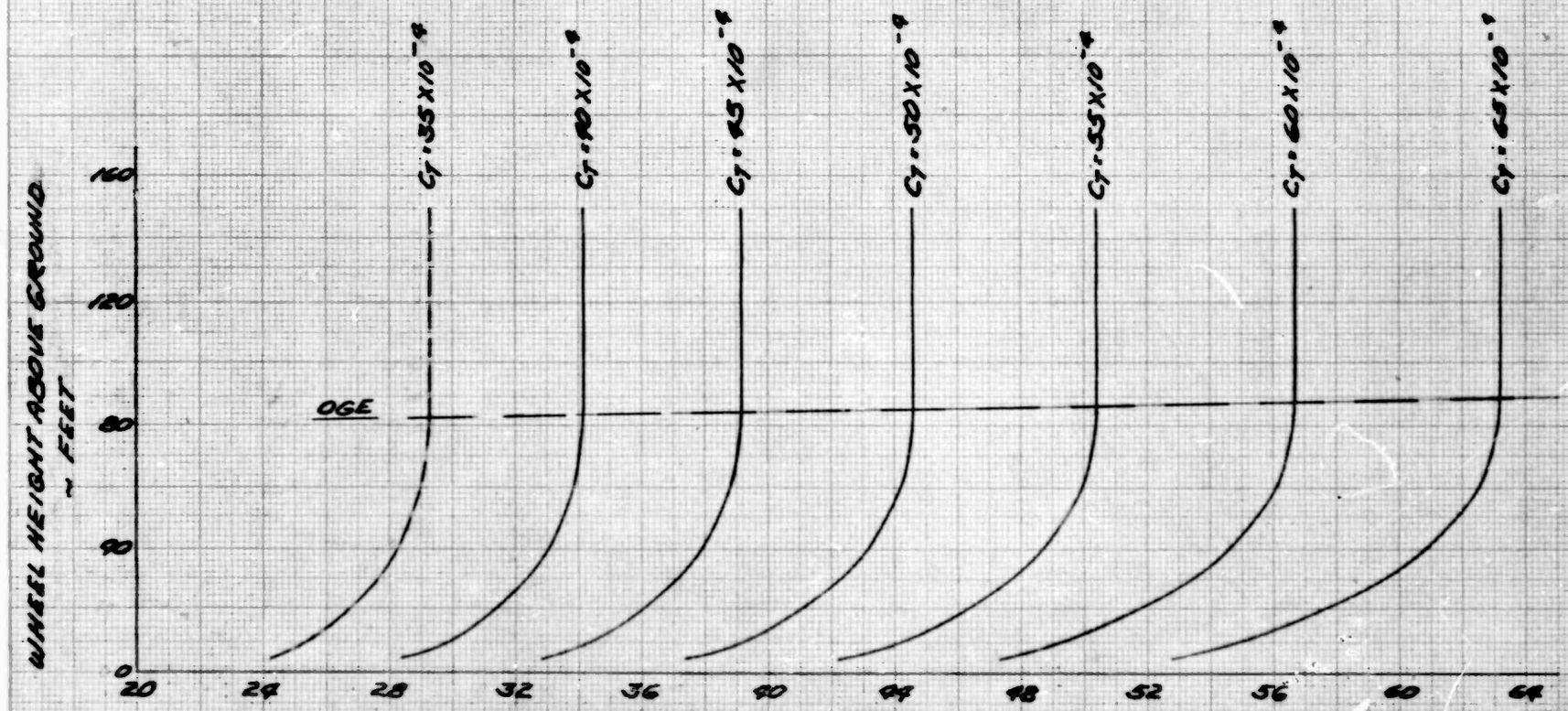
NON-DIMENSIONAL HOVERING PERFORMANCE SUMMARY

CH-47C USA S/N 68-15859

$N_R/\sqrt{\theta} = 200$ RPM

NOTES:

1. CURVES DERIVED FROM FIGURE 7 THROUGH 11.
2. WHEEL HEIGHT MEASURED FROM THE BOTTOM OF THE RIGHT REAR WHEEL.
3. WIND LESS THAN 3 KNOTS.
4. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT



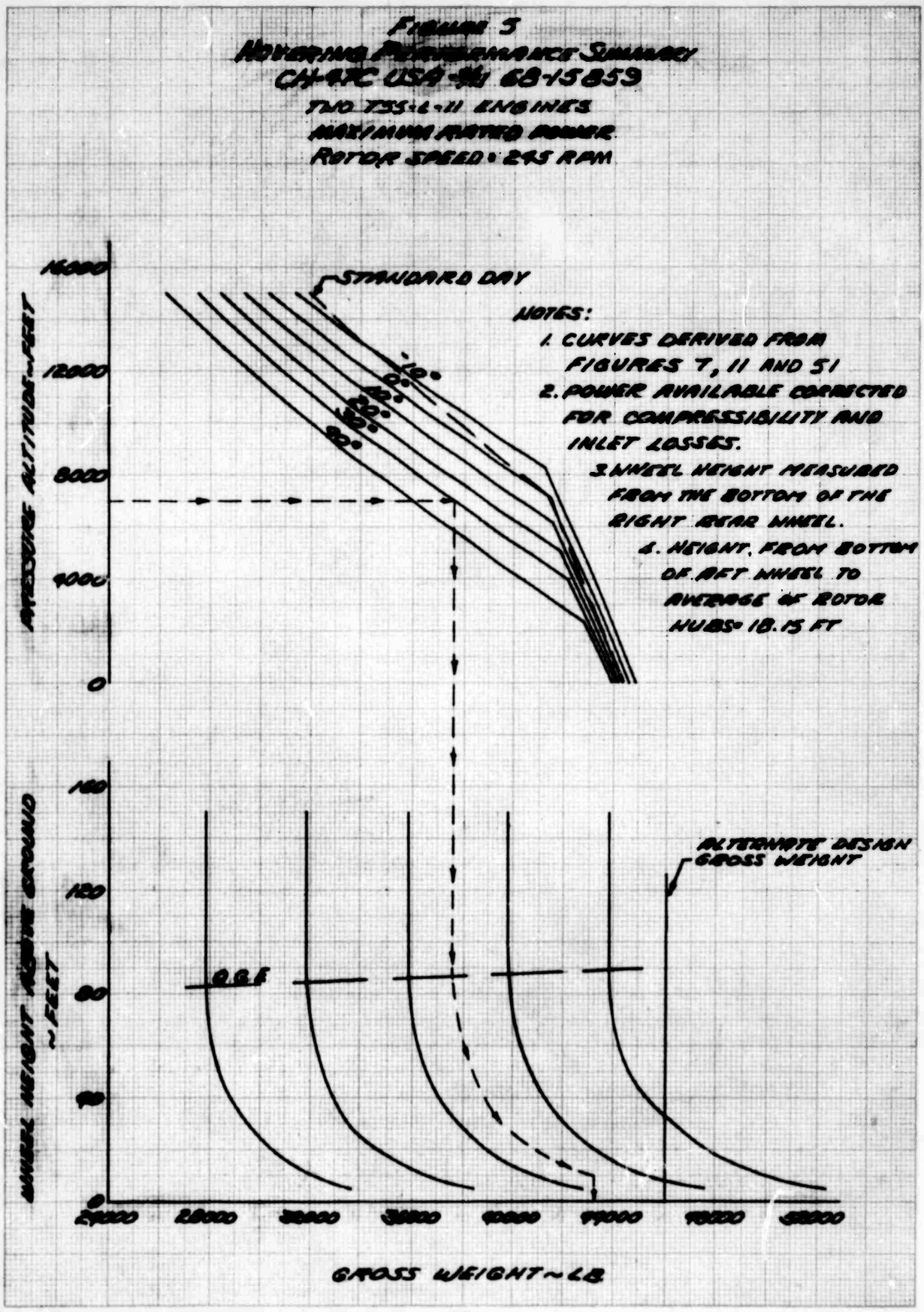


Figure 5
Hovering Performance Summary
CH-47C USA S/N 68-15859
TWO T55-L-11 ENGINES
MAXIMUM RATED POWER
ROTOR SPEED = 245 RPM
STANDARD DAY
NOTES:
1. CURVES DERIVED FROM FIGURES 7, 11 AND 51
2. POWER AVAILABLE CORRECTED FOR COMPRESSIBILITY AND INLET LOSSES.
3. WHEEL HEIGHT MEASURED FROM THE BOTTOM OF THE RIGHT REAR WHEEL.
4. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT
-10°
0°
10°
20°
30°
40°
50°
PRESSURE ALTITUDE ~ FEET
16000
12000
8000
4000
0
WHEEL HEIGHT ABOVE GROUND ~ FEET
160
120
80
40
0
O.G.E.
ALTERNATE DESIGN GROSS WEIGHT
24000
28000
32000
36000
40000
44000
48000
52000
GROSS WEIGHT ~ LB

FIGURE 6

NON-DIMENSIONAL HOVERING PERFORMANCE

CH-47C USA S/N 68-15853

$\Omega_R/\sqrt{\theta} = 200$ RPM

NOTES:

1. CURVES DERIVED FROM FIGURES 7 THROUGH 11.
2. WHEEL HEIGHT MEASURED FROM THE BOTTOM OF THE RIGHT REAR WHEEL.
3. WIND LESS THAN 3 KNOTS.
4. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT

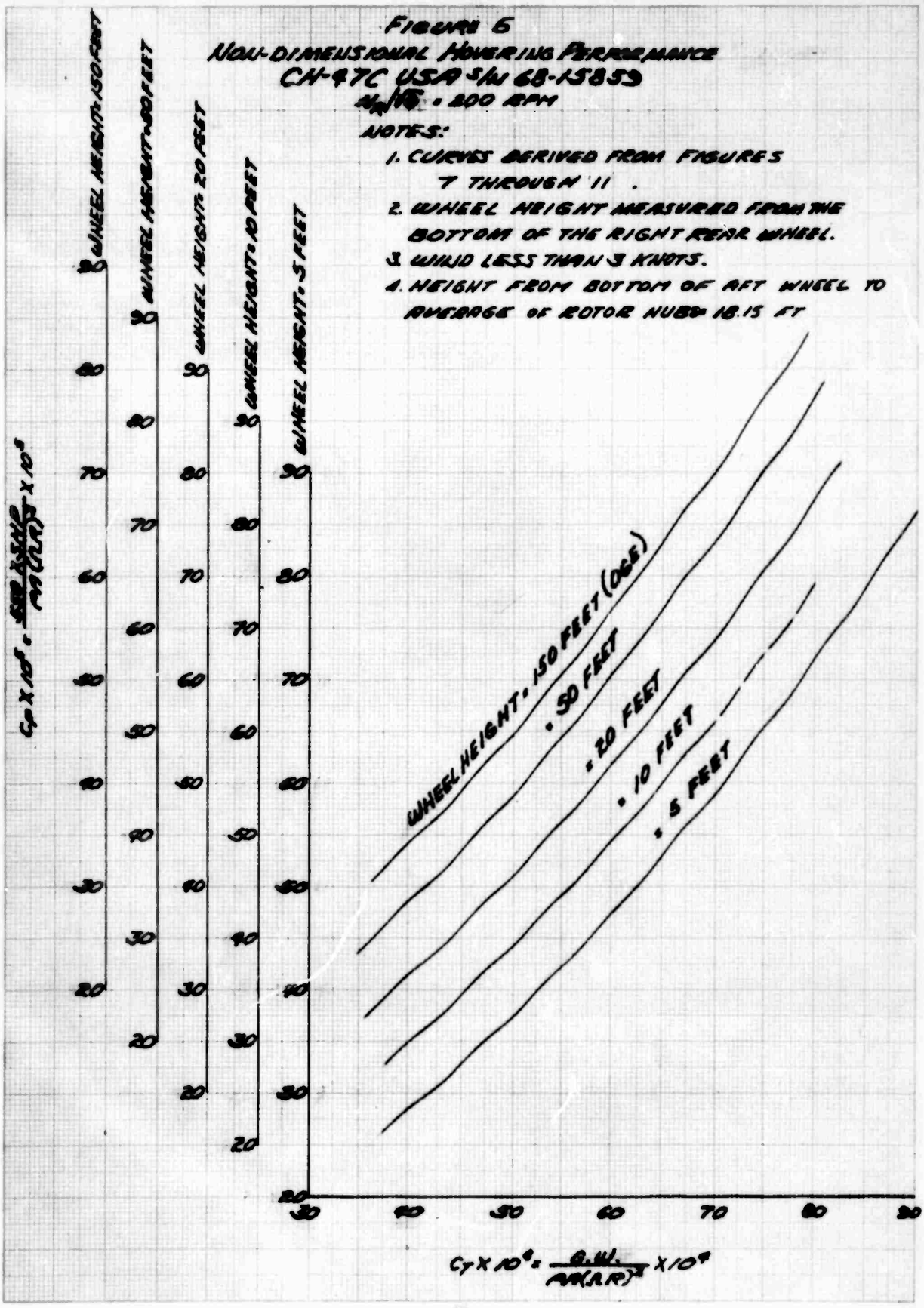


FIGURE 7

NON-DIMENSIONAL HOVERING PERFORMANCE

CH-47C USA 66-15859

5 FOOT MODEL

TETHERED HOVER TESTING

$N_R/\sqrt{\theta_t}$ = 200 RPM

NOTES:

1. WHEEL HEIGHT MEASURED FROM BOTTOM OF RIGHT REAR WHEEL.
2. WIND LESS THAN 3 KNOTS.
3. FIGURE 12 USED TO ELIMINATE ANY COMPRESSIBILITY EFFECTS.
4. ACTUAL ROTOR SPEEDS FROM 214.5 TO 253 RPM.
5. OPEN SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 10,500 FEET.
6. CLOSED SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF ~100 FEET.
7. HEIGHT FROM BOTTOM OF RIGHT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT

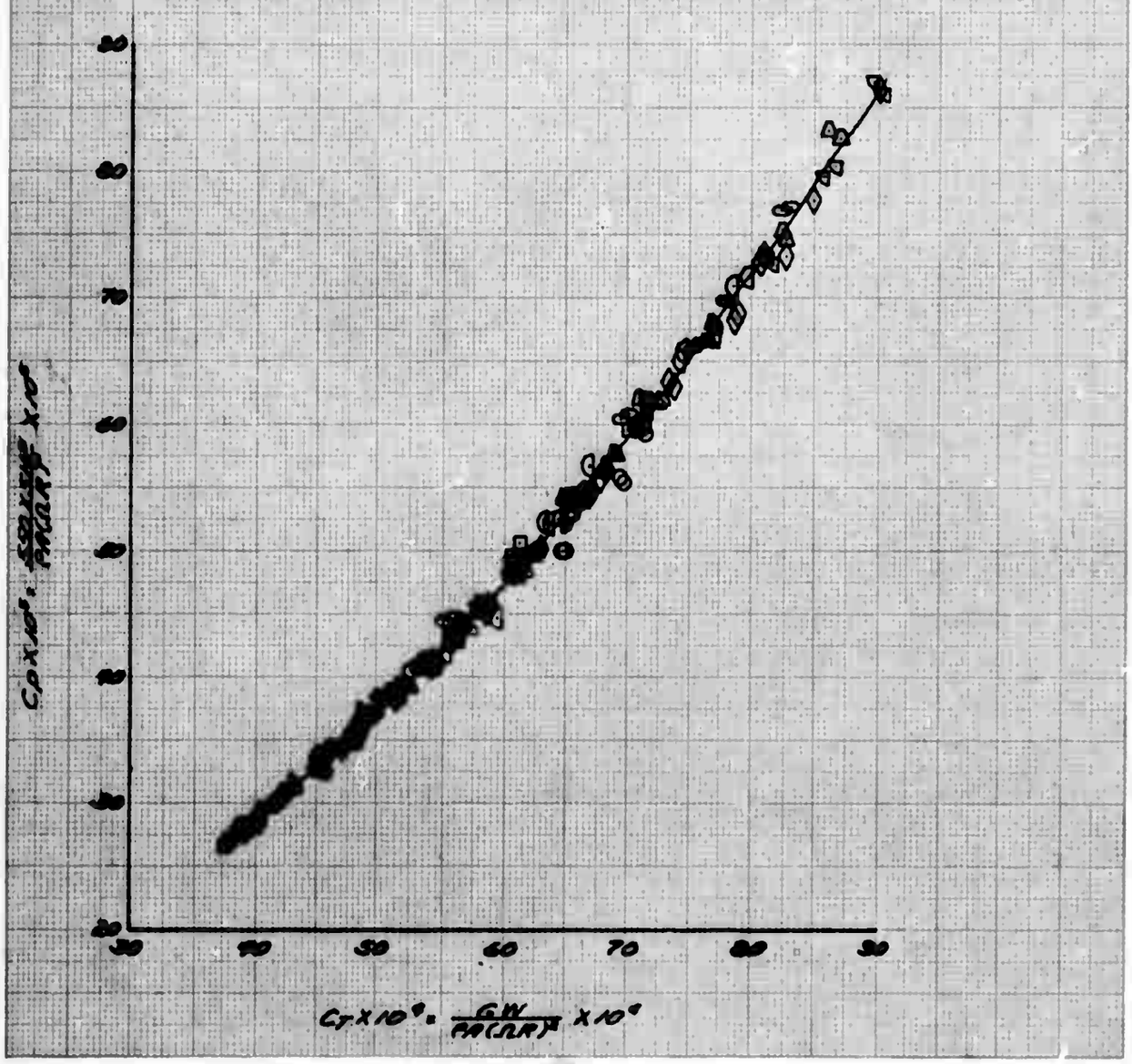


FIGURE 8
NON-DIMENSIONAL HOVERING PERFORMANCE
CH-47C USA S/N 68-15859
10 FOOT HOVER
TETHERED HOVER METHOD
$N_R/\sqrt{\theta} = 200$ RPM

NOTES:

1. WHEEL HEIGHT MEASURED FROM BOTTOM OF RIGHT REAR WHEEL.
2. WIND LESS THAN 3 KNOTS.
3. FIGURE 12 USED TO ELIMINATE ANY COMPRESSIBILITY EFFECTS.
4. ACTUAL ROTOR SPEEDS FROM 227 TO 251.5 RPM.
5. OPEN SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF ~290 FEET
6. CLOSED SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 11130 FEET AND FREE FLIGHT HOVER METHOD.
7. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT

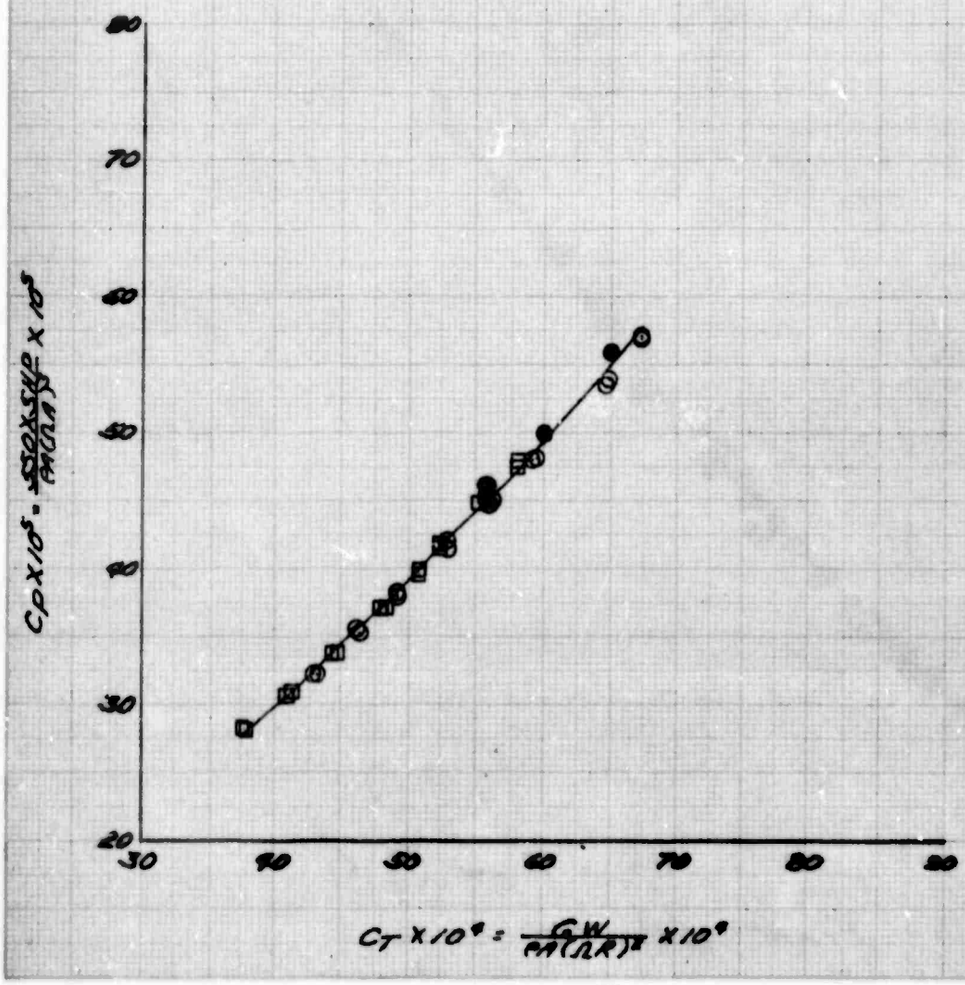


FIGURE 9

NON-DIMENSIONAL HOVERING PERFORMANCE

CH-47C USA S/N 68-15853

20 FOOT HOVER

TETHERED HOVER METHOD

N_R [illegible] = 200 RPM

NOTES:

1. WHEEL HEIGHT MEASURED FROM BOTTOM OF RIGHT REAR WHEEL.
2. WIND LESS THAN 3 KNOTS.
3. FIGURE 12 USED TO ELIMINATE ANY COMPRESSIBILITY EFFECTS.
4. ACTUAL ROTOR SPEEDS FROM 219 TO 250 RPM.
5. OPEN SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 10,950 FEET.
6. CLOSED SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 930 FEET.
7. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS 18.15 FT

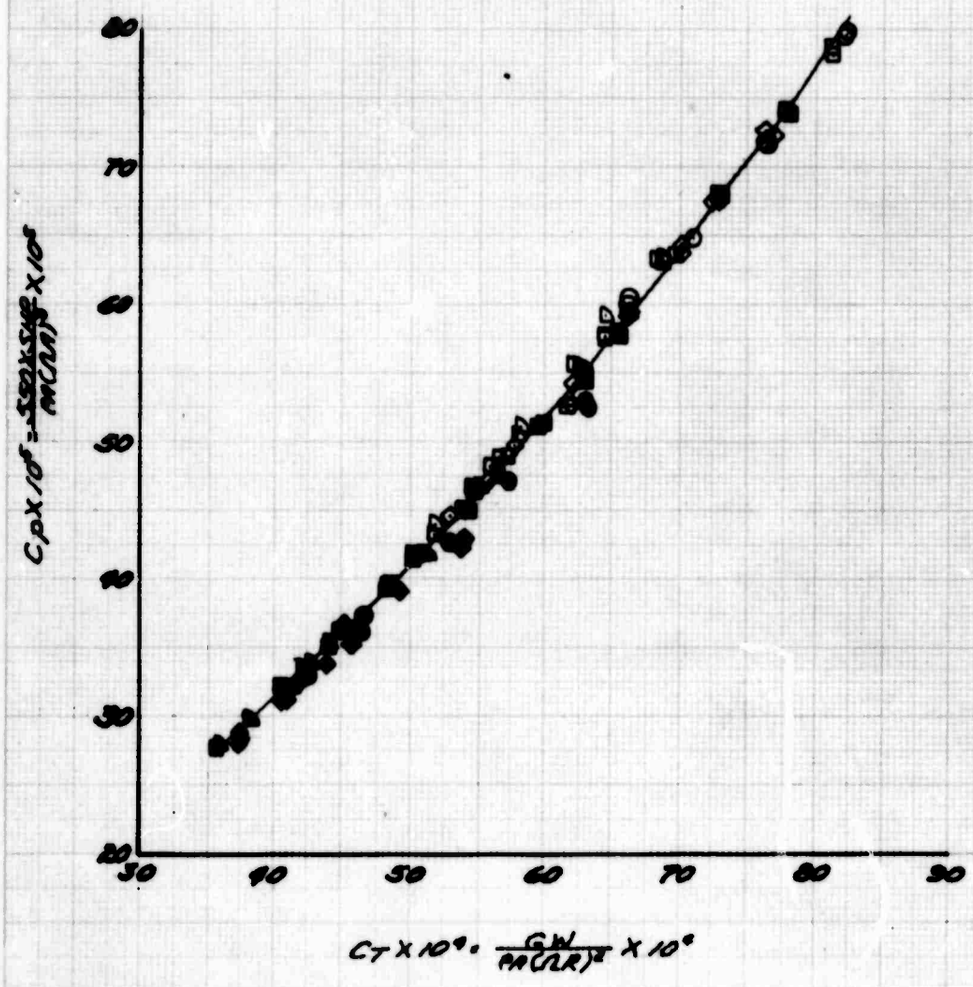


FIGURE 10
NON-DIMENSIONAL HOVERING PERFORMANCE
CH-47C USA S/N 68-15853
30 FOOT HOVER
TETHERED HOVER METHOD
N_R = 200 RPM

NOTES:

1. WHEEL HEIGHT MEASURED FROM BOTTOM OF RIGHT REAR WHEEL.
2. WIND LESS THAN 3 KNOTS.
3. FIGURE 12 USED TO ELIMINATE ANY COMPRESSIBILITY EFFECTS.
4. ACTUAL ROTOR SPEEDS FROM 220 TO 223.5 RPM.
5. OPEN SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 10650 FEET.
6. CLOSED SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 640 FEET.
7. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT

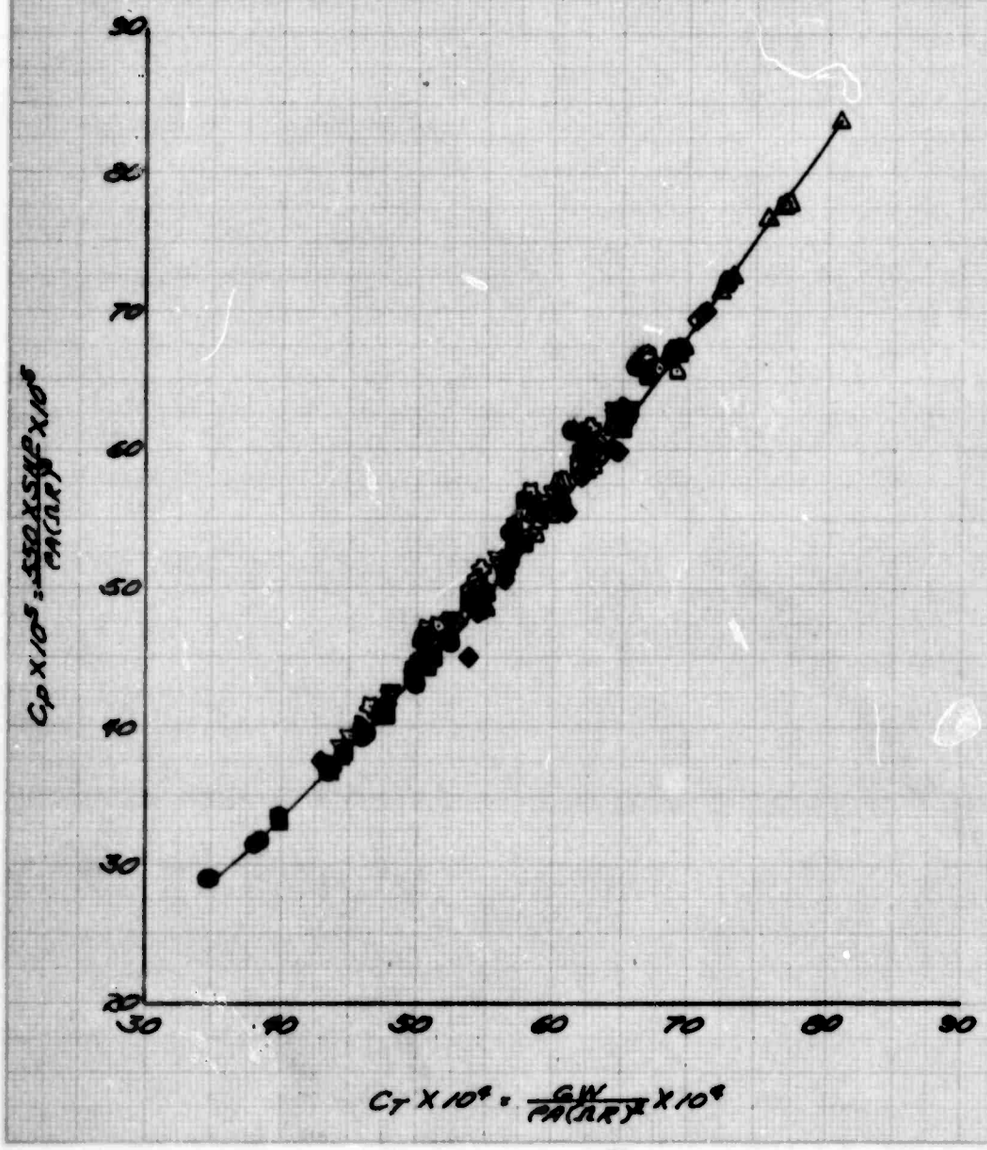


FIGURE 11
NON-DIMENSIONAL HOVERING PERFORMANCE
CH-47C USA S/N 68-15859
150 FOOT HOVER
TETHERED HOVER METHOD
$N_R/\sqrt{\theta} = 200$ RPM

NOTES:

1. WHEEL HEIGHT MEASURED FROM BOTTOM OF RIGHT REAR WHEEL.
2. WIND LESS THAN 3 KNOTS.
3. FIGURE 12 USED TO ELIMINATE ANY COMPRESSIBILITY EFFECTS.
4. ACTUAL ROTOR SPEEDS FROM 217.5 TO 250 RPM.
5. OPEN SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 1600 FEET.
6. CLOSED SYMBOLS DENOTE AVERAGE DENSITY ALTITUDE OF 980 FEET.
7. HEIGHT FROM BOTTOM OF AFT WHEEL TO AVERAGE OF ROTOR HUBS = 18.15 FT

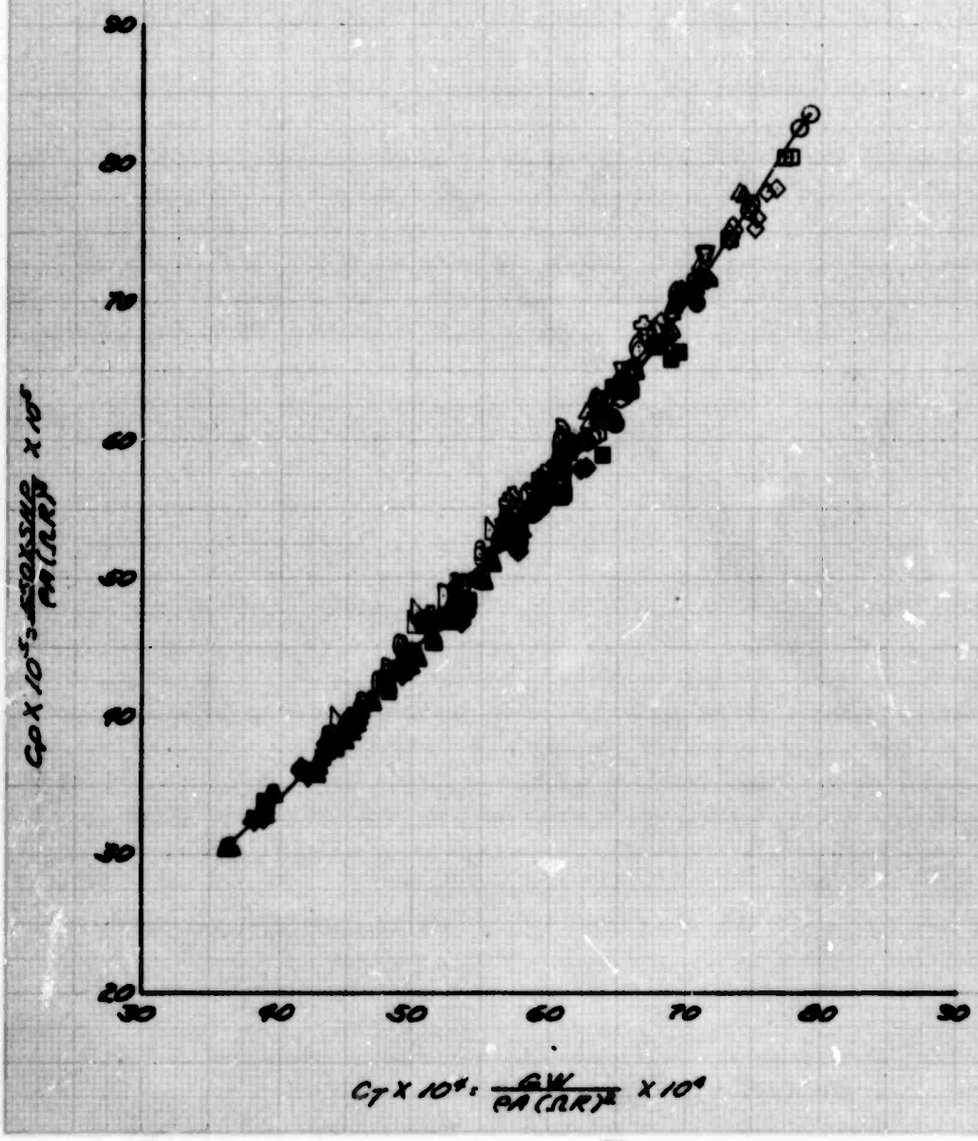
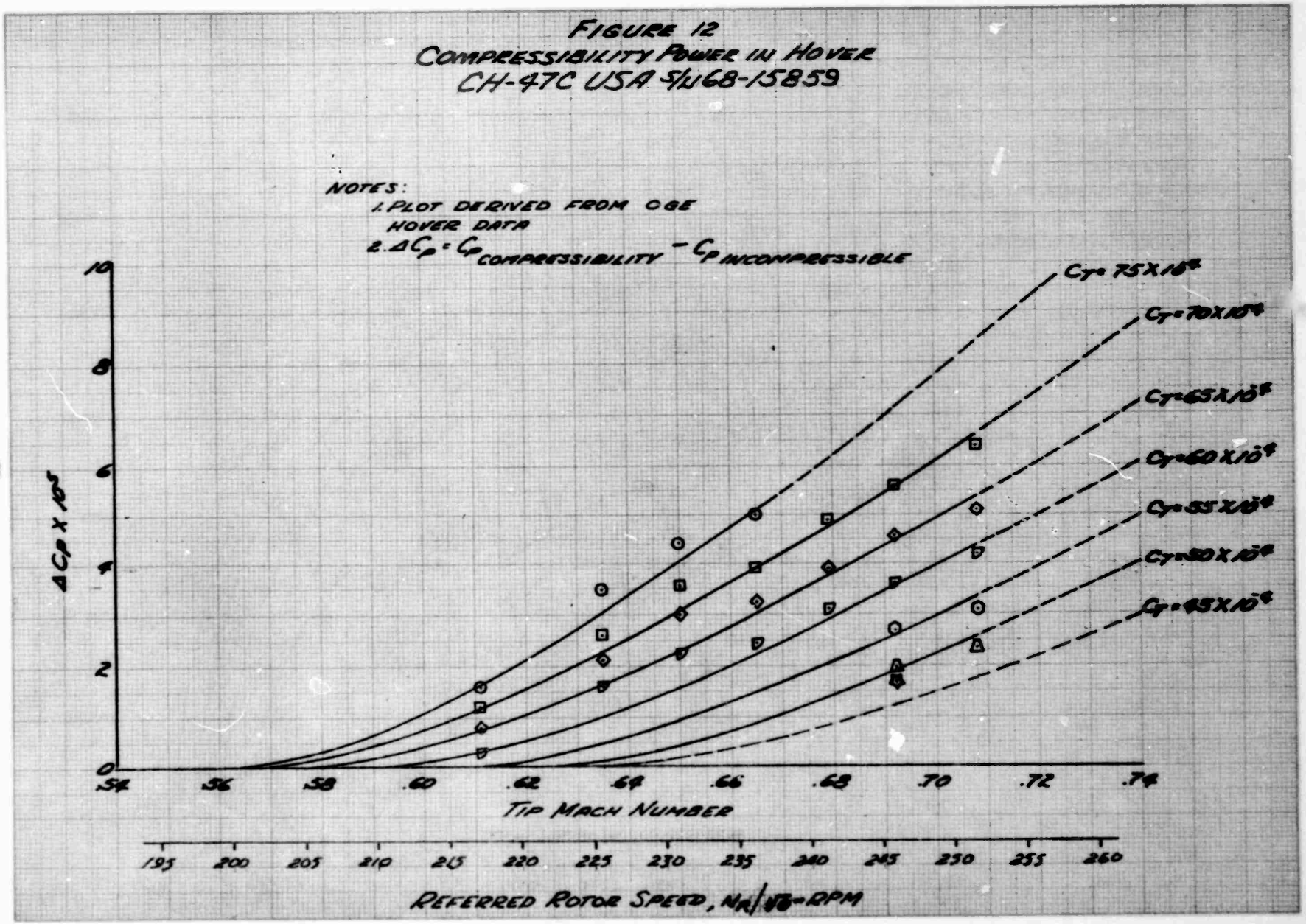


FIGURE 12
COMPRESSIBILITY POWER IN HOVER
CH-47C USA S/N 68-15859



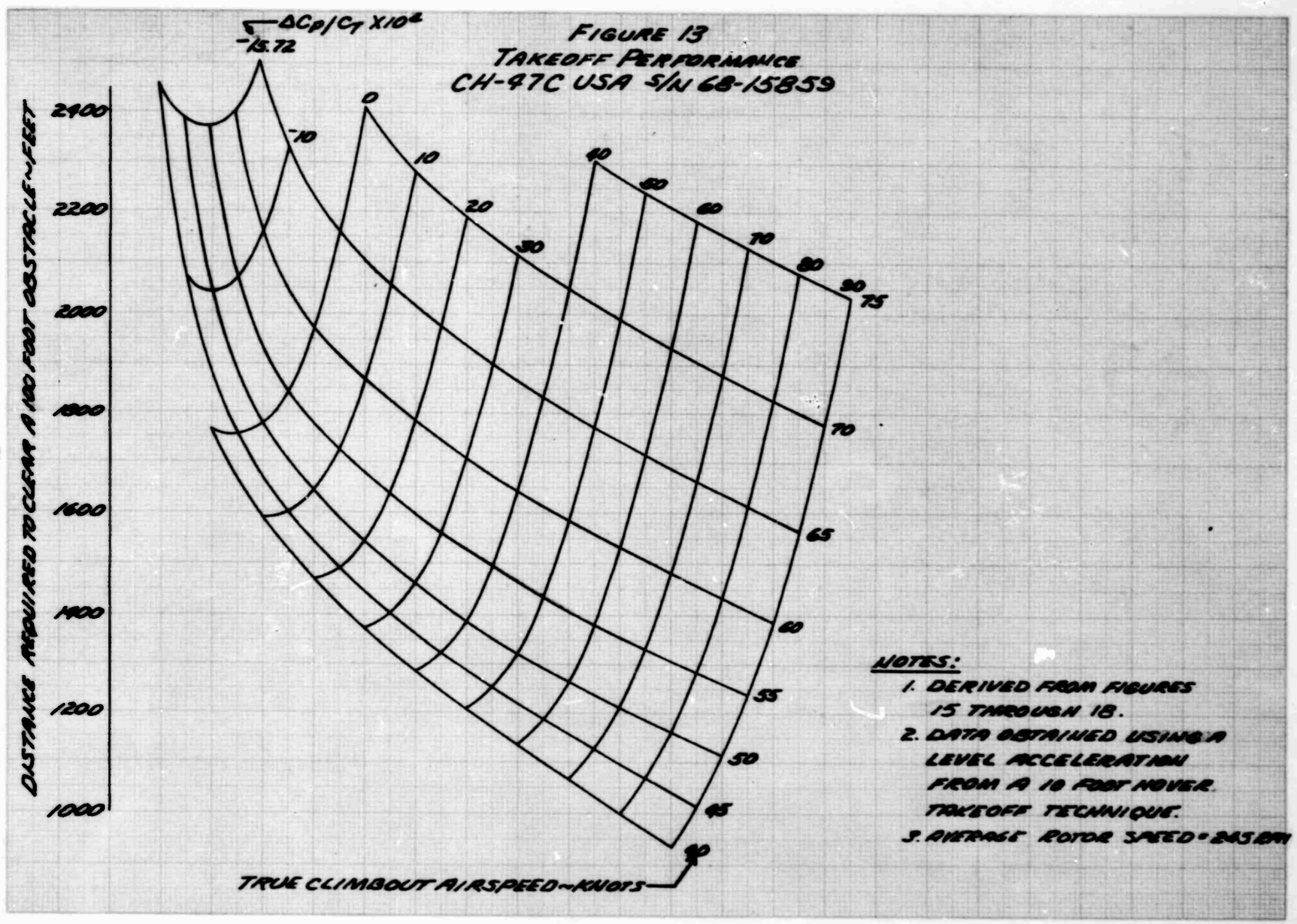


FIGURE 13
TAKEOFF PERFORMANCE
CH-47C USA S/N 68-15859

FIGURE 14
TAKEOFF PERFORMANCE
CH-47C USA S/N 68-15859

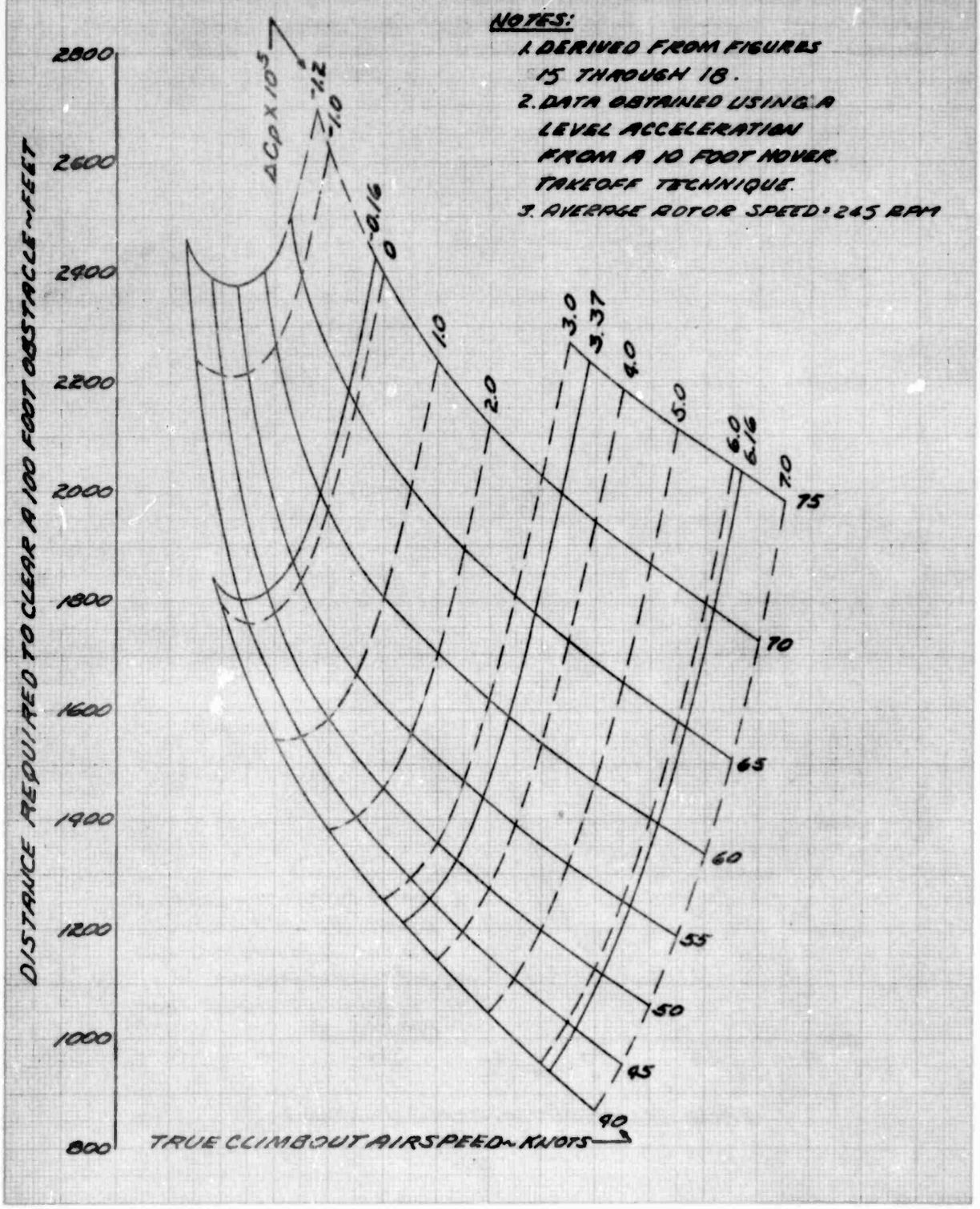


FIGURE 15
TAKEOFF PERFORMANCE
CH-47C USA S/N 68-15853

AVG. GROSS WEIGHT~LB	AVG. PRESSURE ALTITUDE~FT	AVG. OAT ~°C	AVG. ROTOR SPEED~RPM	AVG. C.G. LOCATION~IN	AVG. ΔC_P $\times 10^{5}$	AVG. $\Delta C_P/\sigma_T$ $\times 10^{5}$
33740	3260	13.2	245.5	331.8 (MID)	6.16	88.64

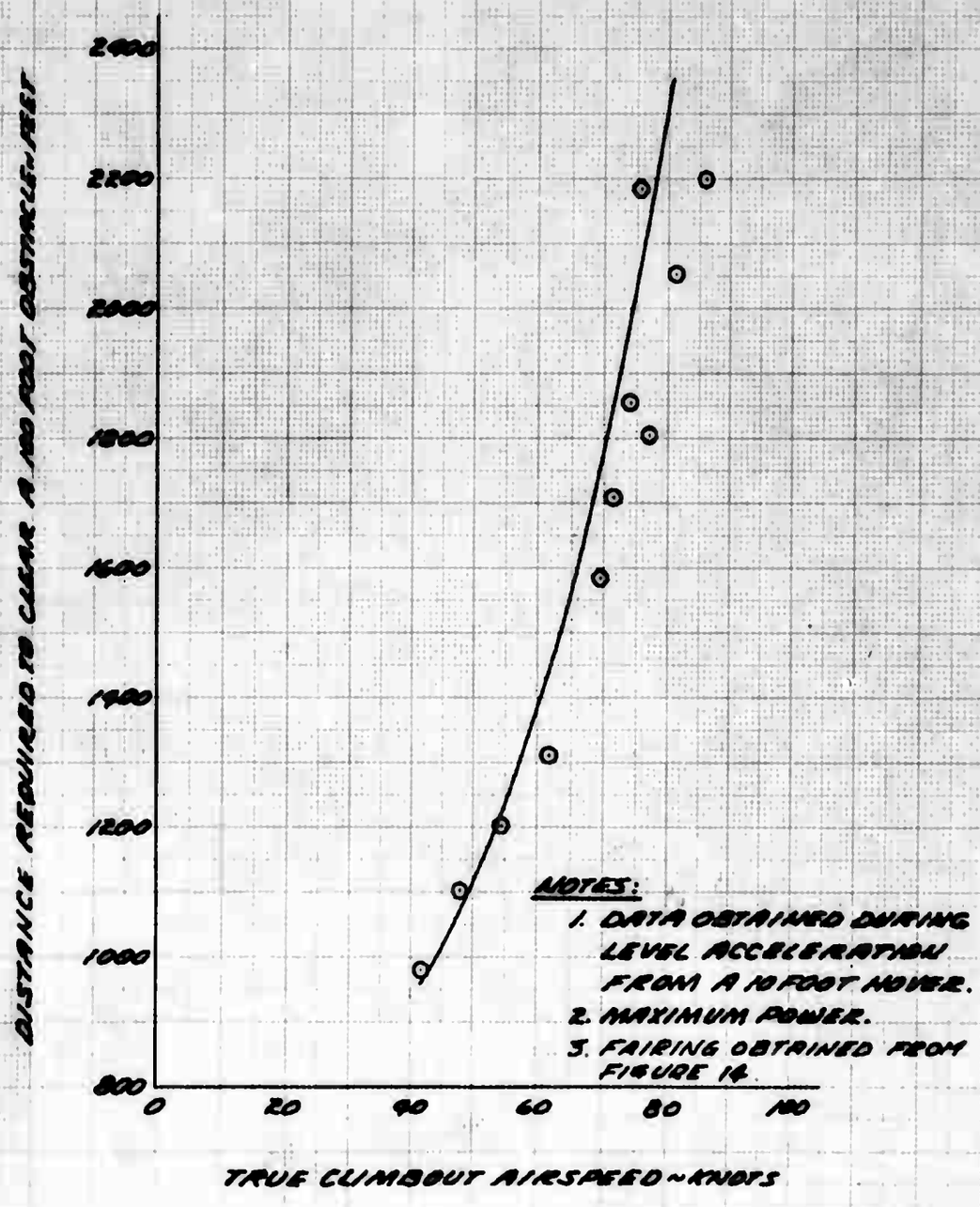


FIGURE 16
TAKEOFF PERFORMANCE
CH-47C USA S/N 68-15859

AVG. GROSS WEIGHT~LB	AVG. PRESSURE ALTITUDE~FT.	AVG. O.A.T. ~°C	AVG. ROTOR SPEED~RPM	AVG. C.G. LOCATION~IN.	AVG. ΔC_p $\times 10^5$	AVG. $\Delta C_p/\sigma_T$ $\times 10^5$
39920	3840	11.0	245.0	330.5 (MID)	3.37	49.7

NOTES:
1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 10 FOOT HOVER.
2. MAXIMUM POWER.
3. FAIRING OBTAINED FROM FIGURE 14

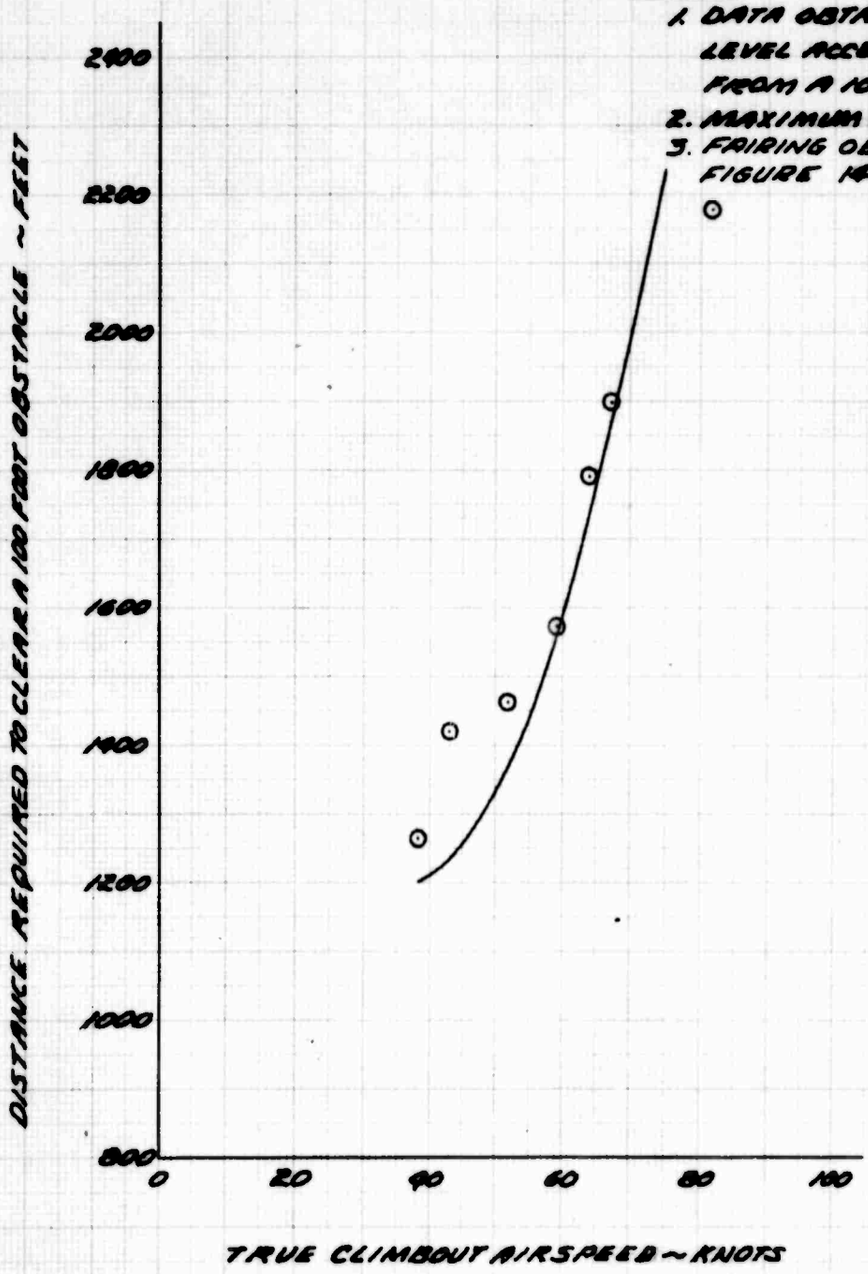


FIGURE 17
TAKEOFF PERFORMANCE
CH-47C USA S/N 68-15859

AVG. GROSS WEIGHT ~ LB	AVG. PRESSURE ALTITUDE ~ FT.	AVG. OAT ~ °C	AVG. ROTOR SPEED ~ RPM	AVG. C.G. LOCATION ~ IN.	AVG. ΔC_p $\times 10^5$	AVG. $\Delta C_p/C_T$ $\times 10^6$
42770	9310	13.0	245.5	331.7 (MID)	-0.16	-2.13

NOTES:
1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 10 FOOT HOVER.
2. MAXIMUM POWER.
3. FAIRING OBTAINED FROM FIGURE 14

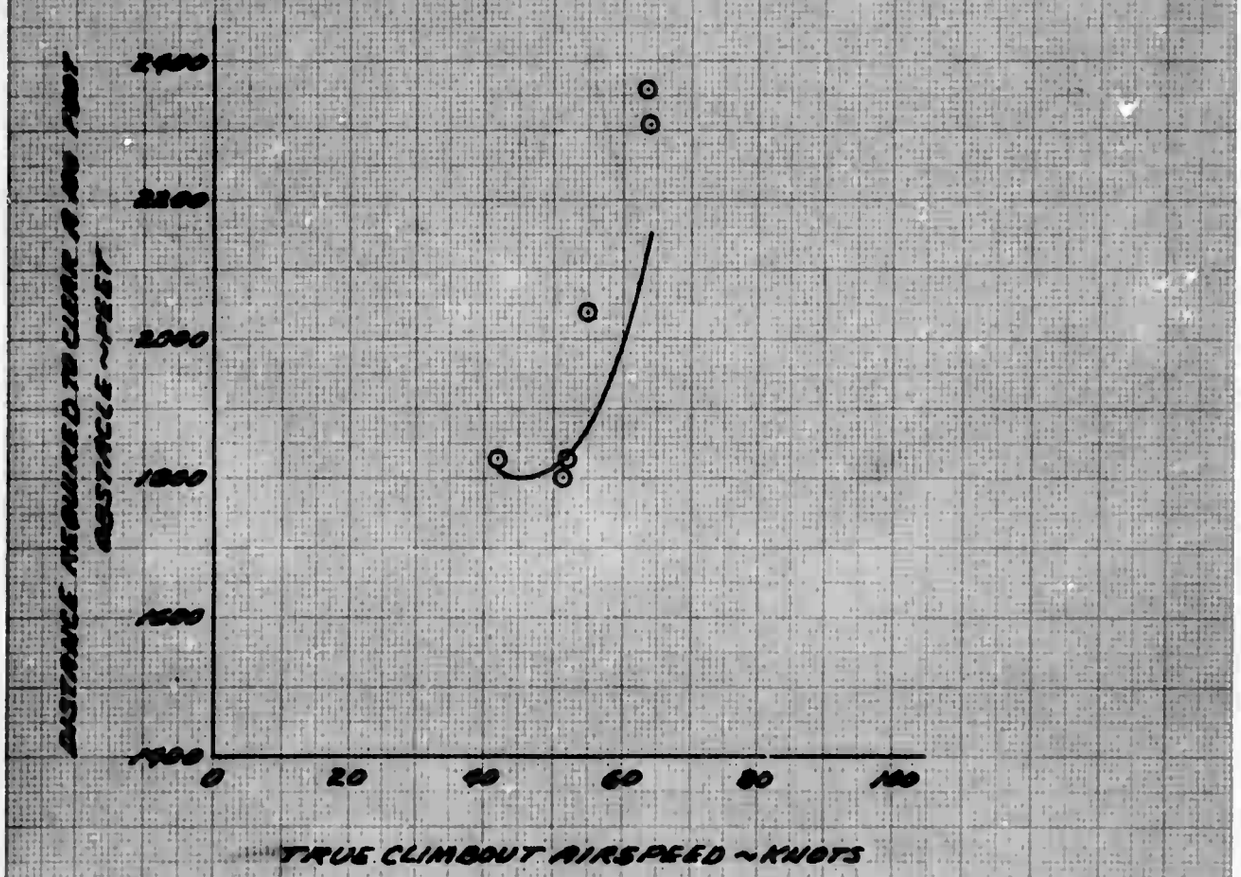


FIGURE 18
TAKEOFF PERFORMANCE
CH-47C USA S/N 68-15859

AVG. GROSS WEIGHT~LB	AVG. PRESSURE ALTITUDE~FT.	AVG. OAT ~°C	AVG. ROTOR SPEED~RPM	AVG. C.G. LOCATION~IN.	AVG. ΔC_p $\times 10^3$	AVG. $\Delta C_p/C_T$ $\times 10^5$
43960	9330	7.9	245.0	331.7 (MID)	-1.2	-15.72

NOTES:
1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 10 FOOT HOVER.
2. MAXIMUM POWER.
3. FAIRING OBTAINED FROM FIGURE 14

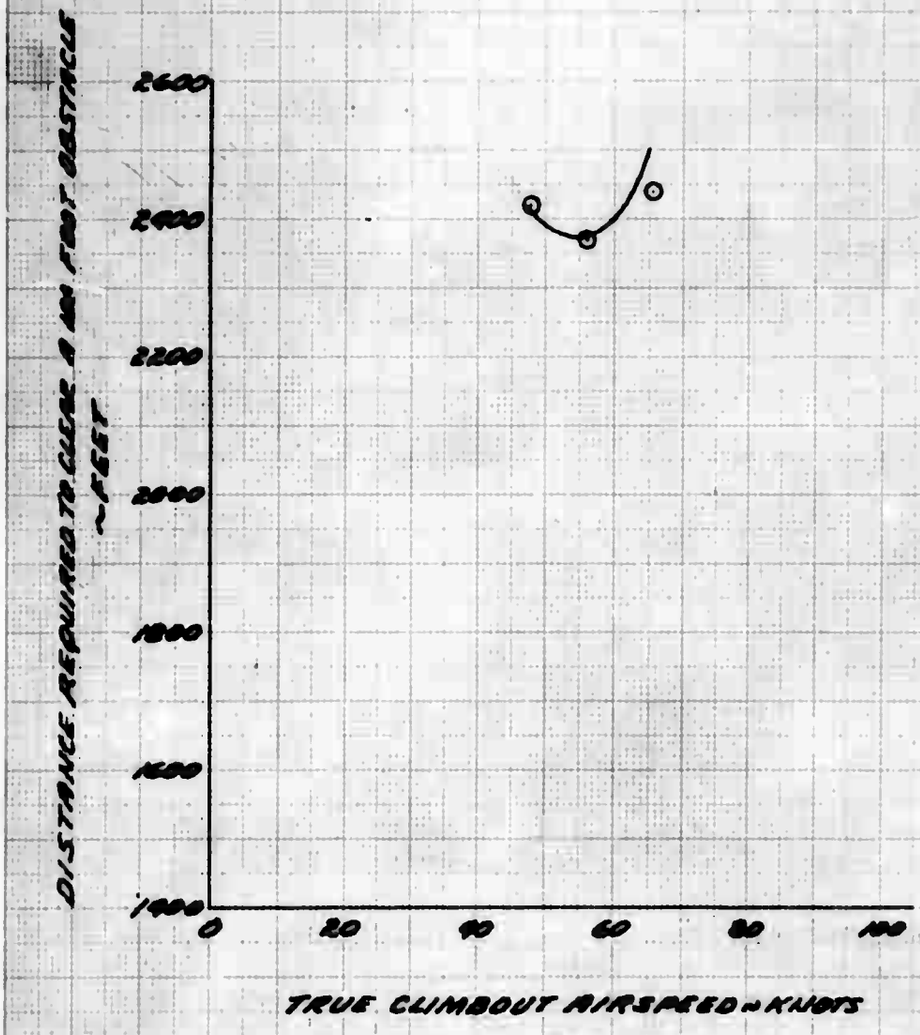


FIGURE 19
CLIMB PERFORMANCE
CH-47C USA S/N 68-15859
SINGLE ENGINE
STANDARD DAY

MILITARY RATED POWER — ROTOR SPEED = 230 RPM
CLIMB START GROSS WEIGHT = 37,365 LB — C.G. LOCATION = MID
POWER AVAILABLE OBTAINED FROM ENGINE MODEL
SPECIFICATION NO. 124.27 R WITH RAM CORRECTION.

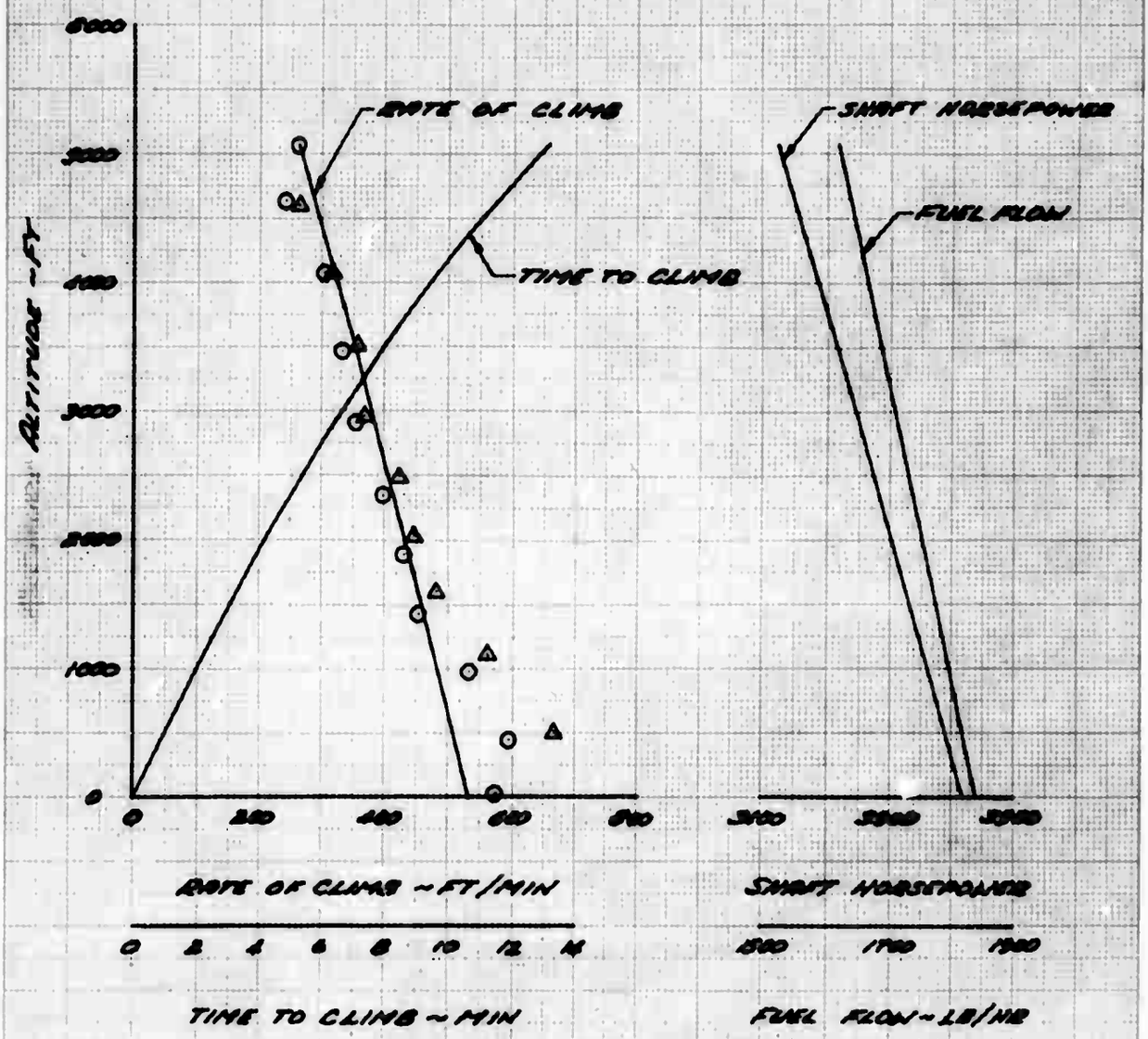
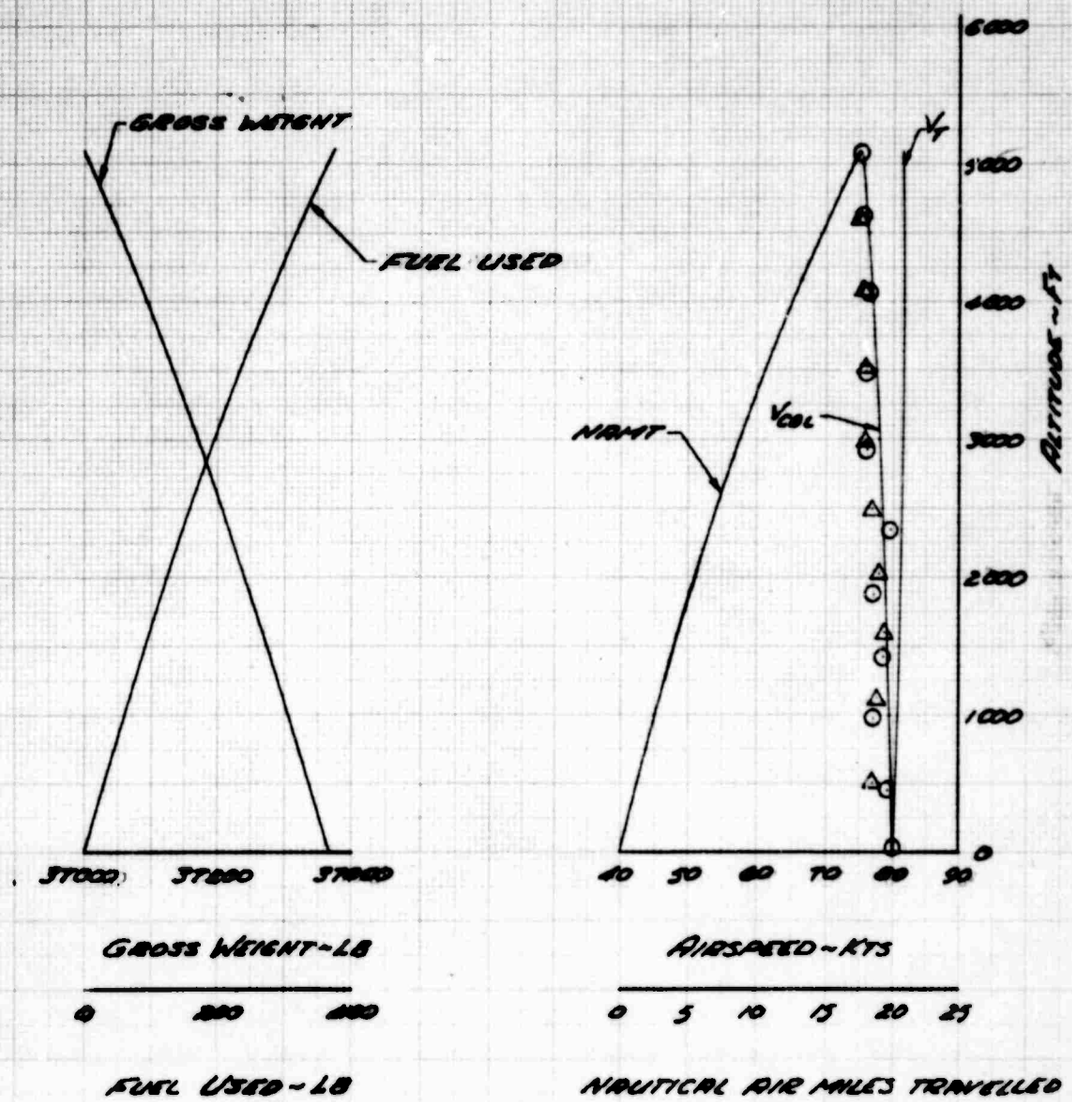


FIGURE 19
CLIMB PERFORMANCE (CONTINUED)
CH-47C USA S/N 68-15859



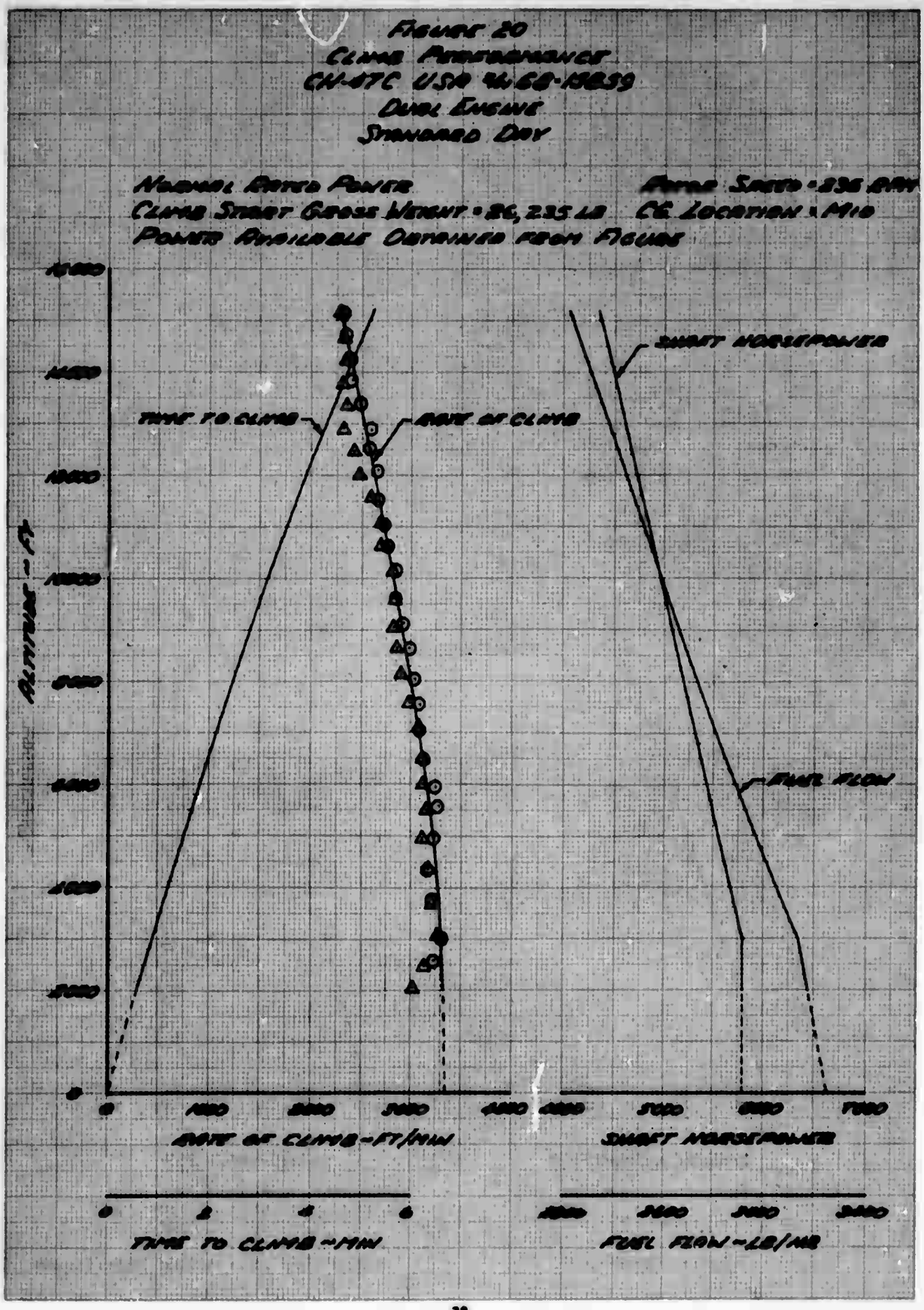
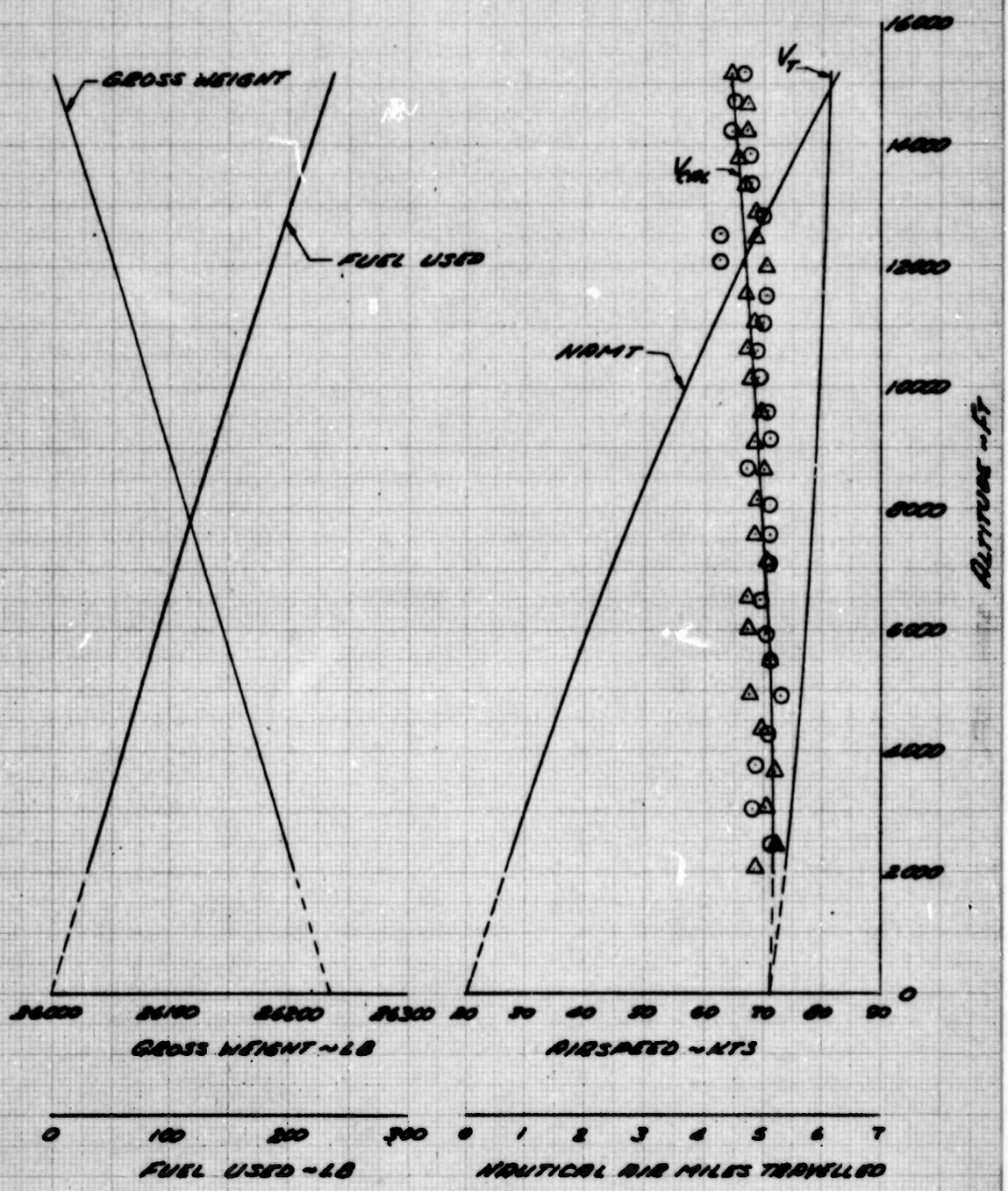


FIGURE 20
CLIMB PERFORMANCE
CH-47C USA 66-19839
DUAL ENGINE
STANDARD DAY
NORMAL RATED POWER
CLIMB START GROSS WEIGHT = 26,235 LB
POWER AVAILABLE OBTAINED FROM FIGURE
CG LOCATION = M10
SHAFT HORSEPOWER
TIME TO CLIMB
RATE OF CLIMB
FUEL FLOW
ALTITUDE ~ FT
RATE OF CLIMB ~ FT/MIN
SHAFT HORSEPOWER
TIME TO CLIMB ~ MIN
FUEL FLOW ~ LB/HR

FIGURE 20
CLIMB PERFORMANCE (CONTINUED)
CH-47C USA S/N 68-15859



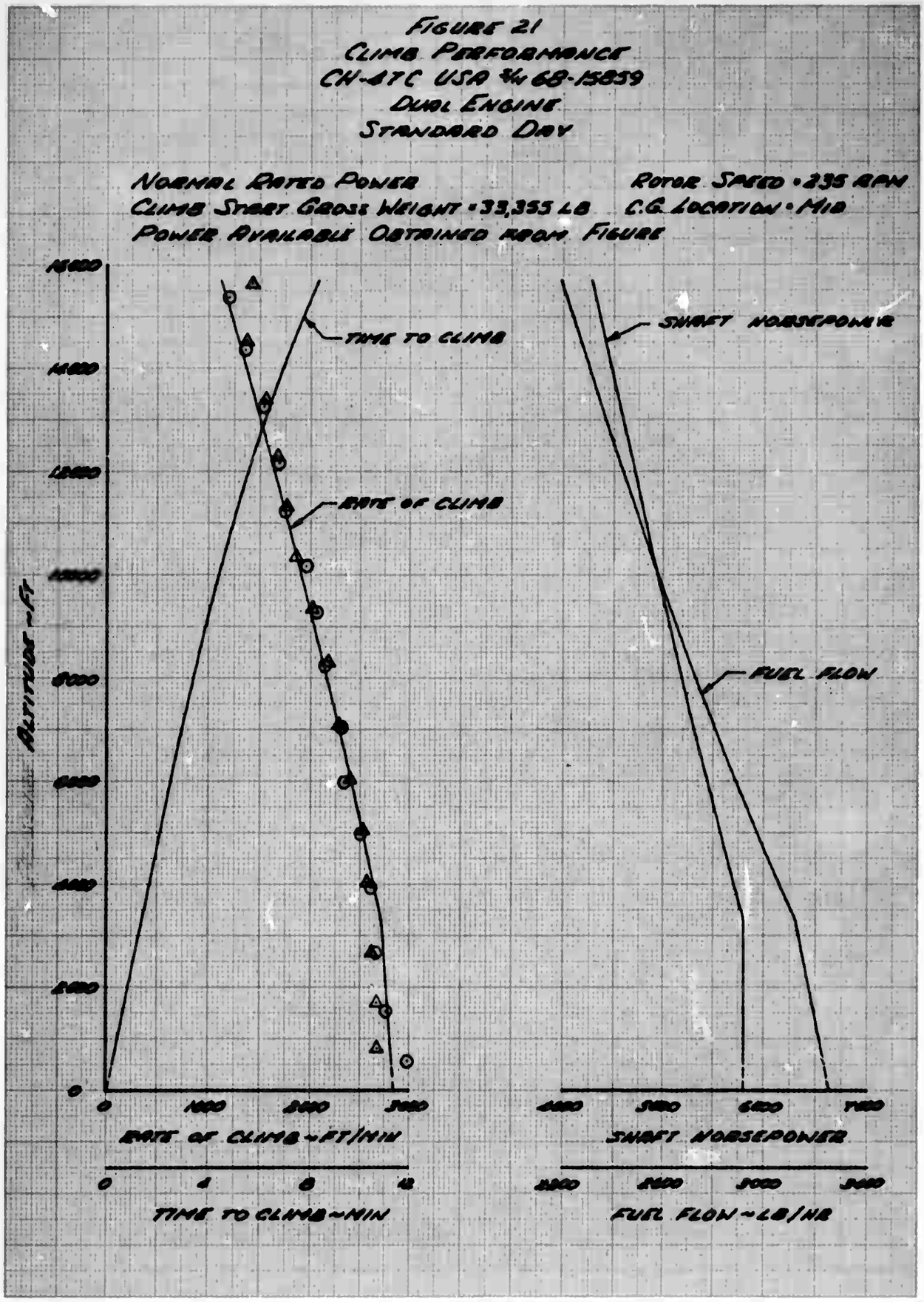


FIGURE 21
CLIMB PERFORMANCE
CH-47C USA S/N 68-15859
DUAL ENGINE
STANDARD DAY
NORMAL RATED POWER
CLIMB START GROSS WEIGHT = 33,355 LB
POWER AVAILABLE OBTAINED FROM FIGURE
ROTOR SPEED = 235 RPM
C.G. LOCATION = MID
TIME TO CLIMB
RATE OF CLIMB
SHAFT HORSEPOWER
FUEL FLOW
ALTITUDE ~ FT
RATE OF CLIMB ~ FT/MIN
TIME TO CLIMB ~ MIN
SHAFT HORSEPOWER
FUEL FLOW ~ LB/HR

FIGURE 21
CLIMB PERFORMANCE (CONTINUED)
CH-47C USA 66-13859

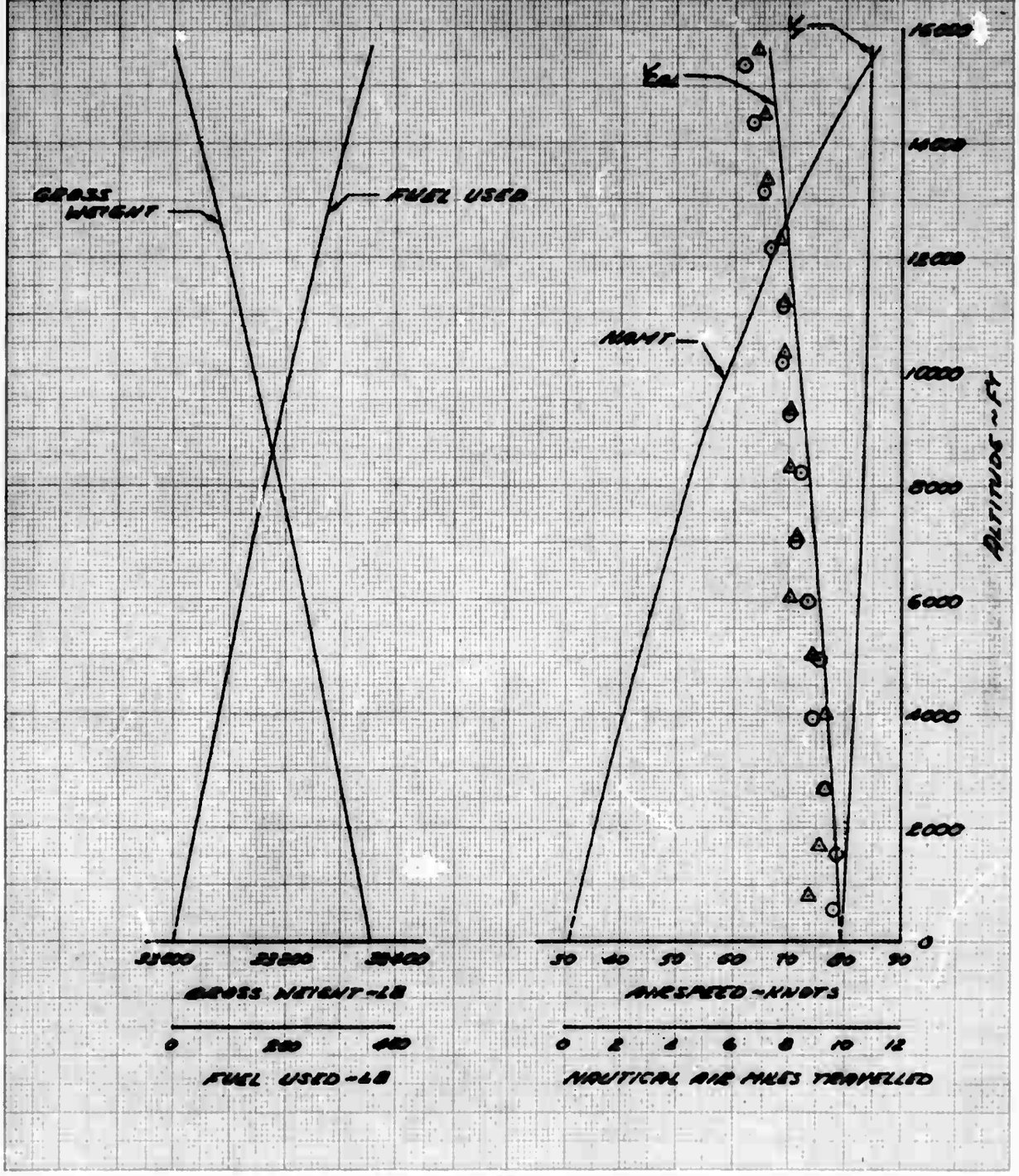


FIGURE 22
CLIMB PERFORMANCE
CH-47C USA S/N 68-15859
DUAL ENGINE
STANDARD DAY

NORMAL RATED POWER — ROTOR SPEED = 245 RPM
CLIMB START GROSS WEIGHT = 46,795 LB — C.G. LOCATION = MID
POWER AVAILABLE OBTAINED FROM FIGURE

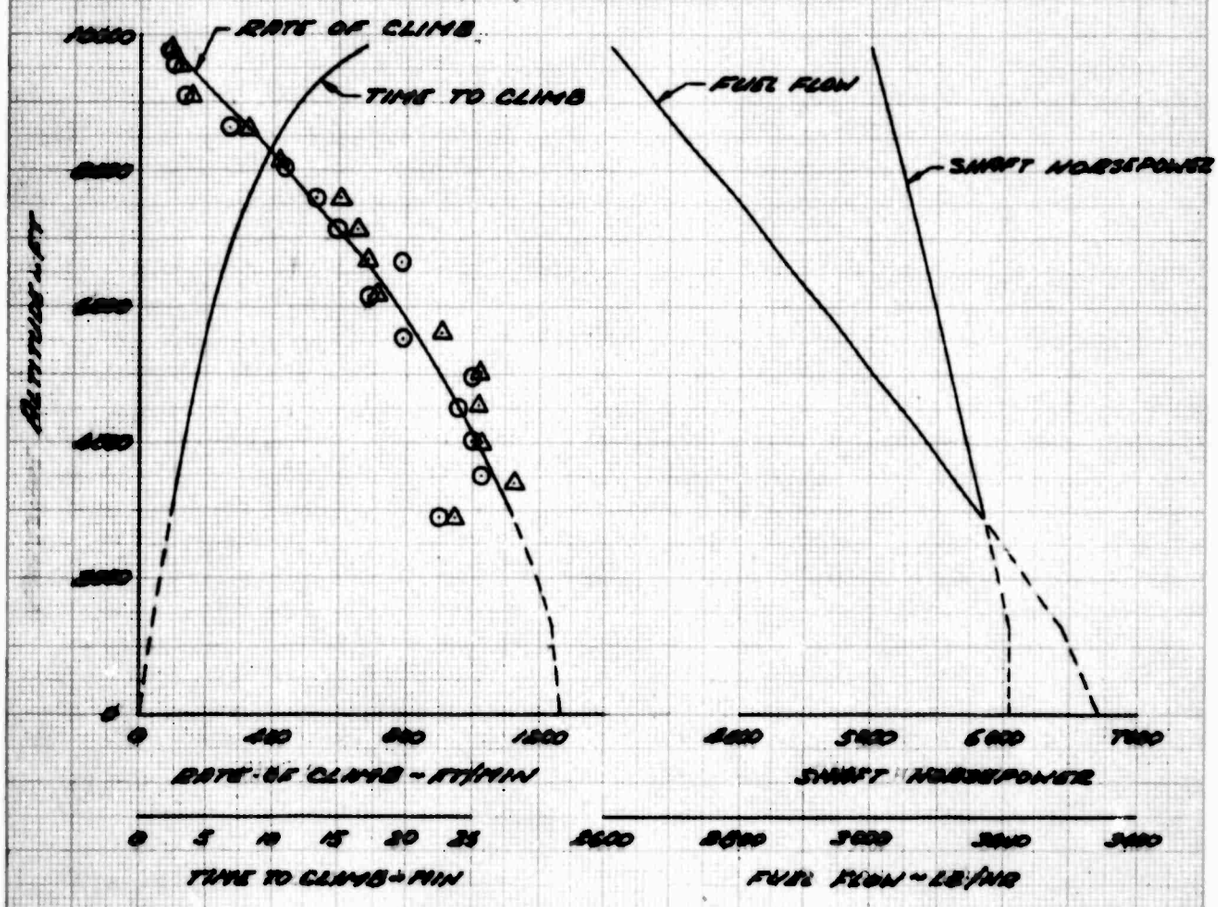
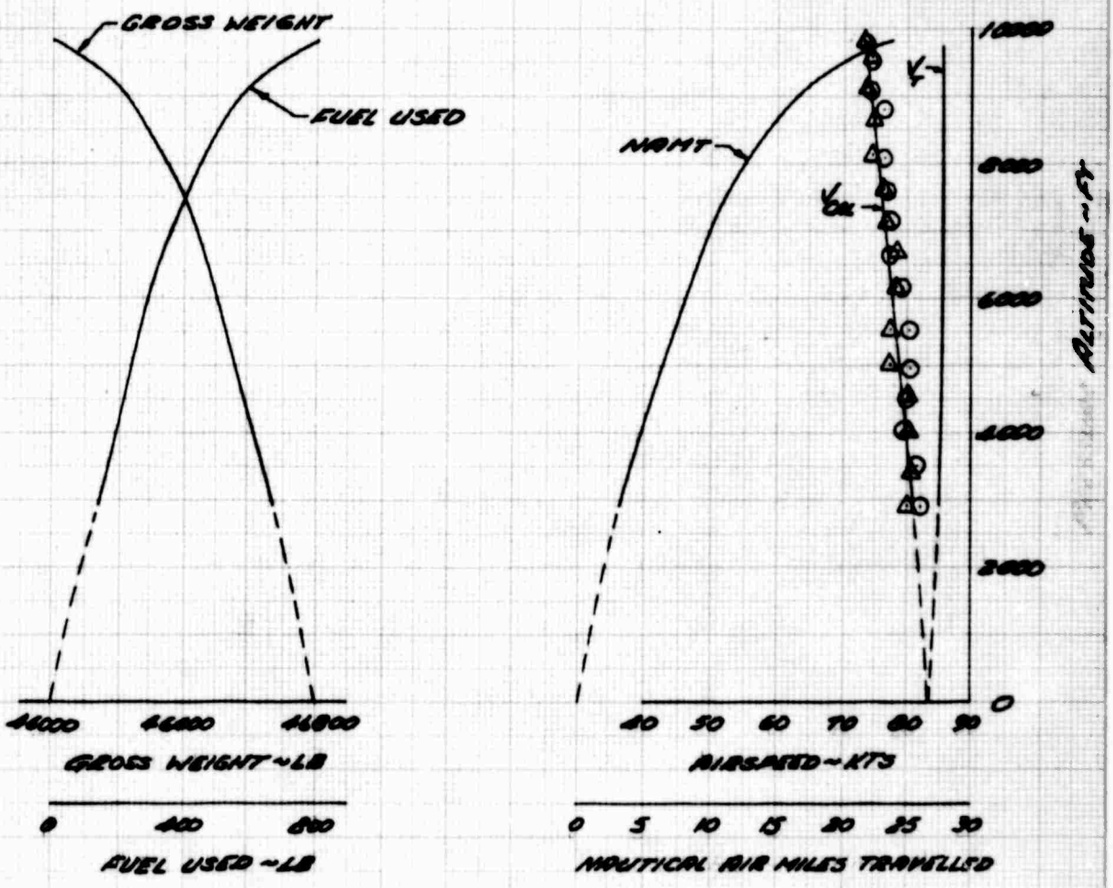


FIGURE 22
CLIMB PERFORMANCE (CONTINUED)
CH-47C USA S/N 68-15859



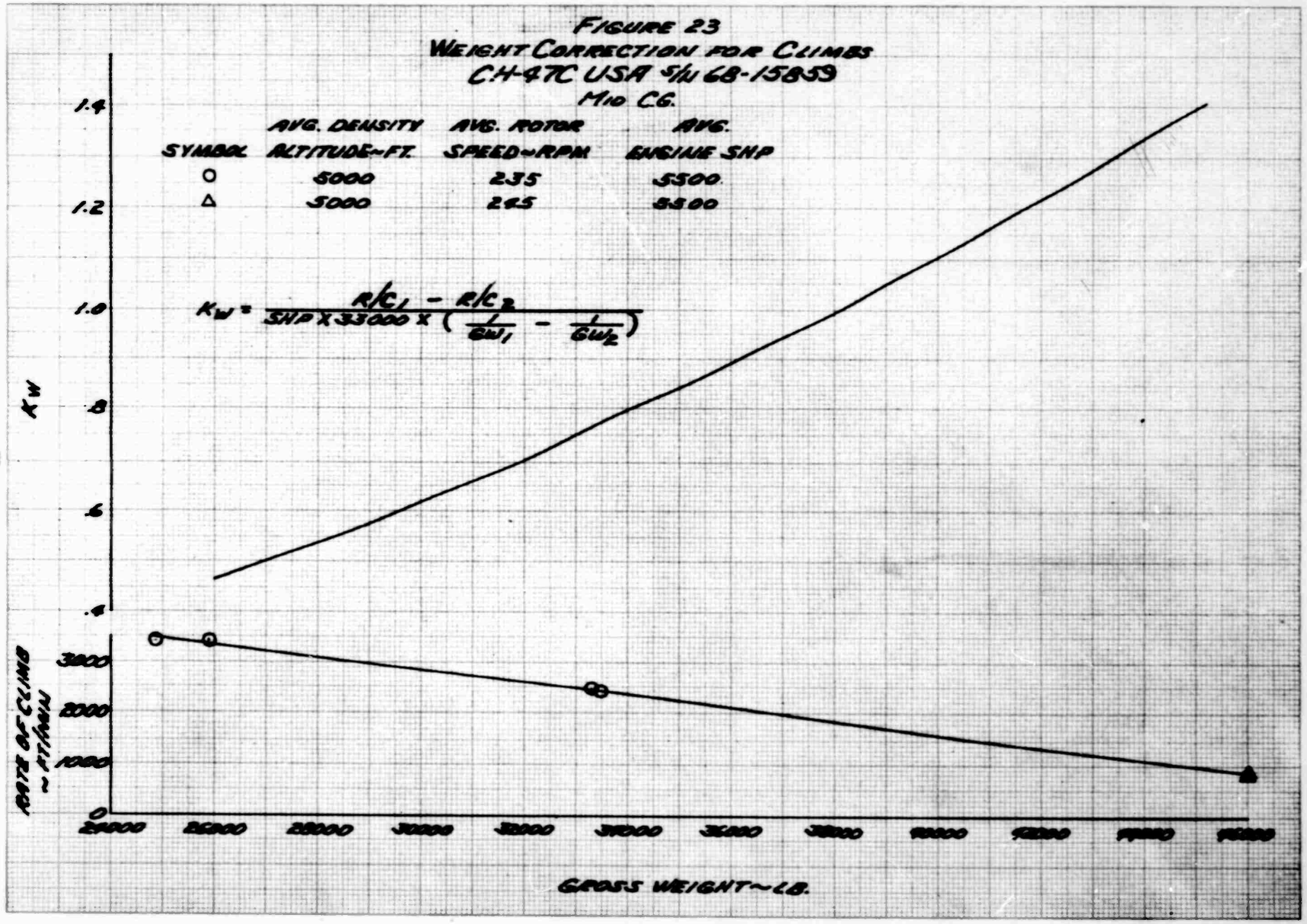


FIGURE 23
WEIGHT CORRECTION FOR CLIMBS
CH-47C USA S/N 68-15859
MID C.G.
SYMBOL
AVG. DENSITY ALTITUDE~FT.
AVG. ROTOR SPEED~RPM
AVG. ENGINE SHP
5000
235
5500
5000
245
5500
$K_W = \frac{R/C_1 - R/C_2}{SHP \times 33000 \times \left(\frac{1}{GW_1} - \frac{1}{GW_2}\right)}$
K_W
1.4
1.2
1.0
.8
.6
.4
RATE OF CLIMB ~ FT/MIN
3000
2000
1000
0
20000
25000
30000
35000
40000
45000
50000
GROSS WEIGHT~LB.

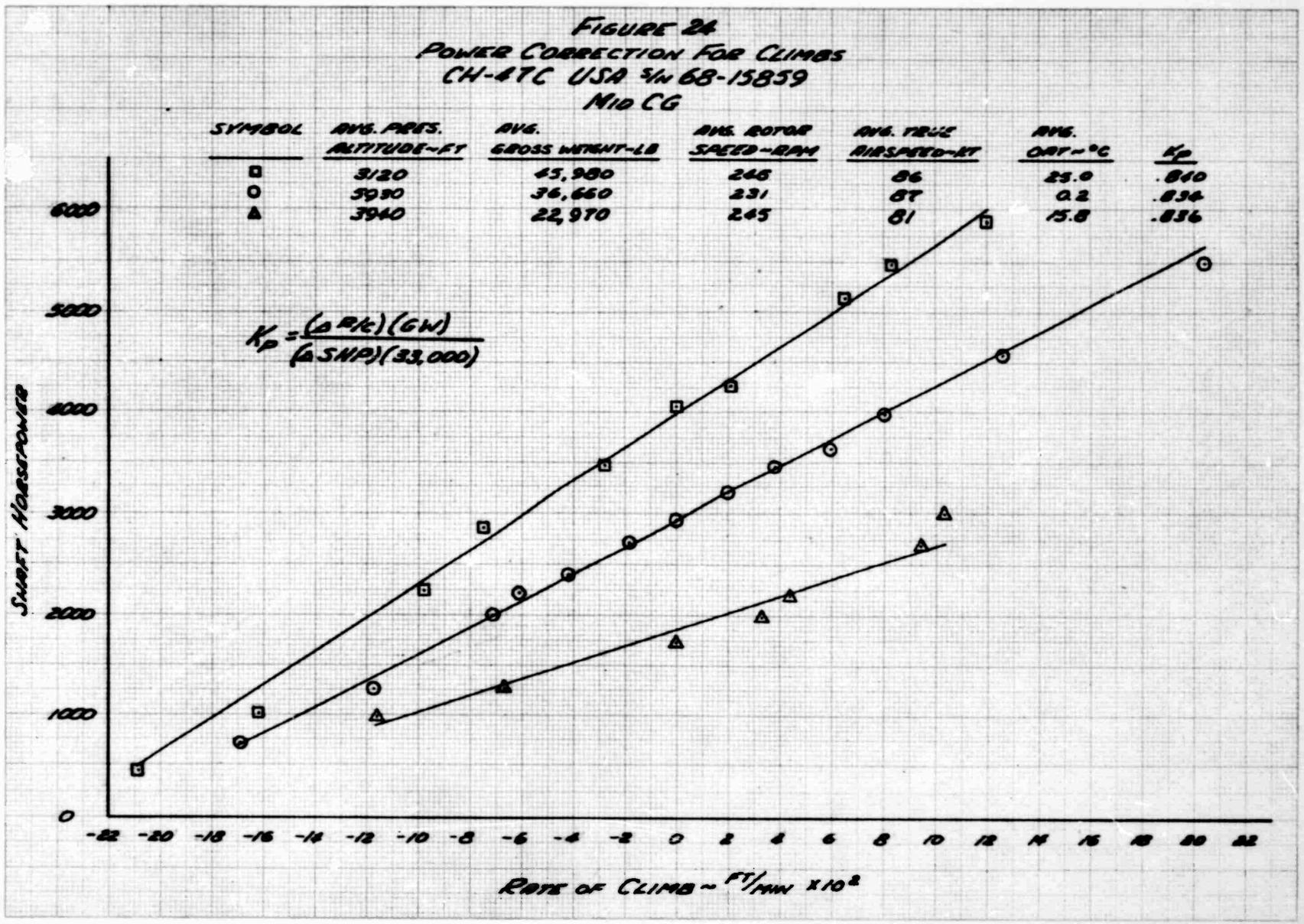


FIGURE 24
POWER CORRECTION FOR CLIMBS
CH-47C USA S/N 68-15859
MID CG
SYMBOL
AVG. PRES. ALTITUDE~FT
AVG. GROSS WEIGHT~LB
AVG. ROTOR SPEED~RPM
AVG. TRUE AIRSPEED~KT
AVG. OAT~°C
Kp
3120 45,980 246 86 25.0 .840
5930 36,660 231 87 0.2 .834
3940 22,970 245 81 15.8 .836
$K_p = \frac{(\Delta R/C)(GW)}{(\Delta SHP)(33,000)}$
SHAFT HORSEPOWER
0
1000
2000
3000
4000
5000
6000
-22 -20 -18 -16 -14 -12 -10 -8 -6 -4 -2 0 2 4 6 8 10 12 14 16 18 20 22
RATE OF CLIMB ~ FT/MIN X10²

FIGURE 25
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 22.5 RPM

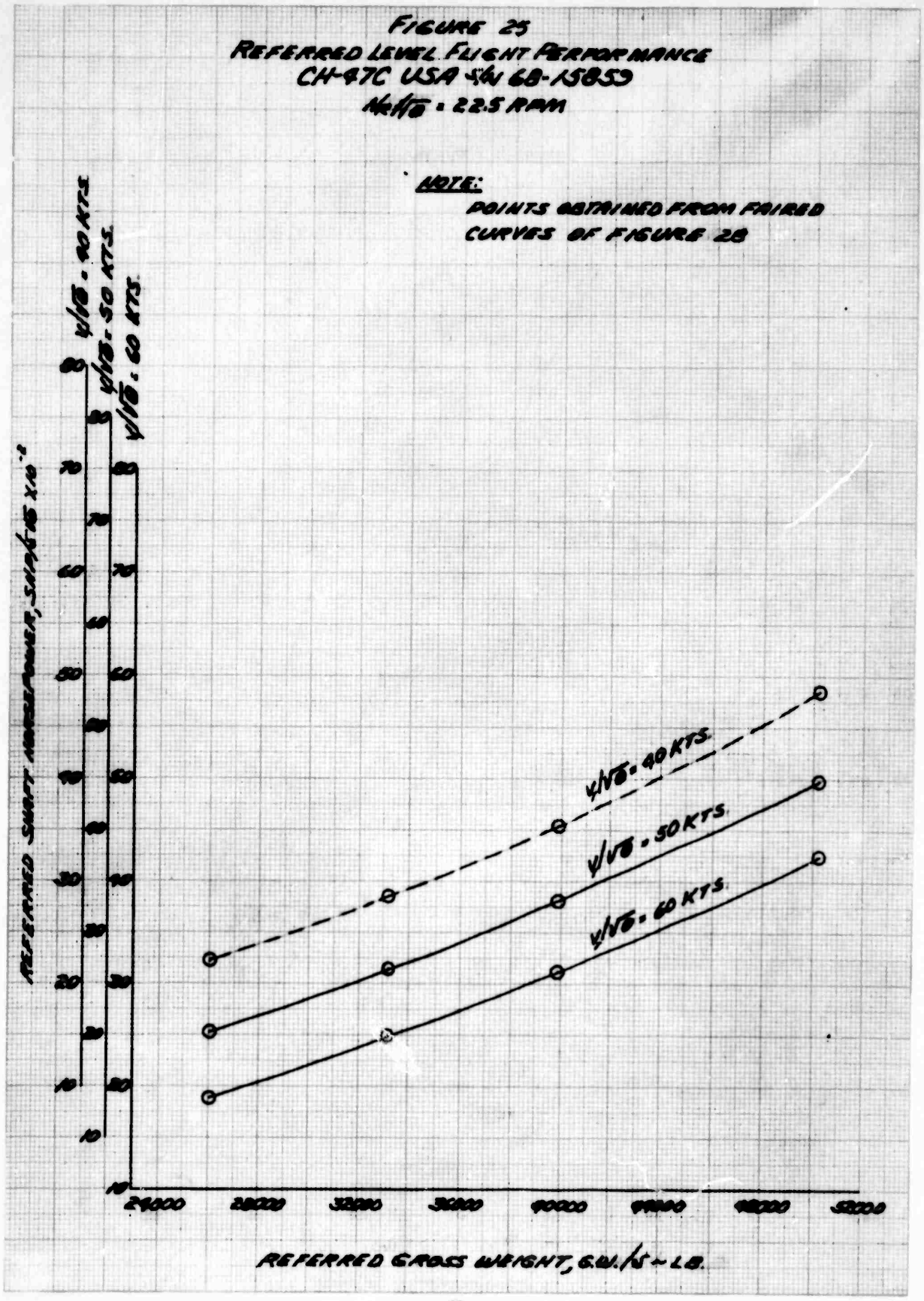


FIGURE 26
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_2/\sqrt{\theta}$ = 225 RPM

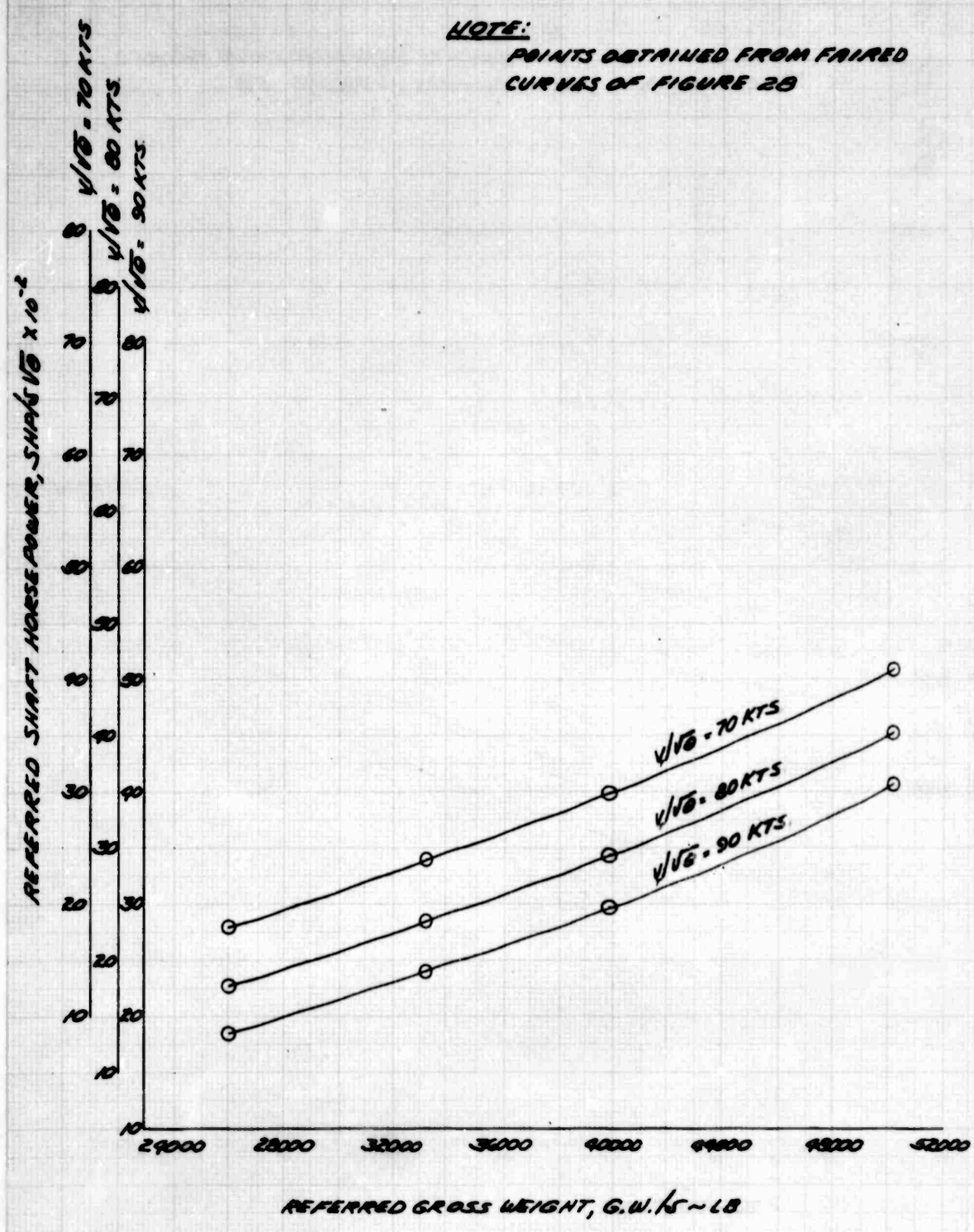


FIGURE 27
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 225 RPM

NOTE:
POINTS OBTAINED FROM FAIRED CURVES OF FIGURE 28

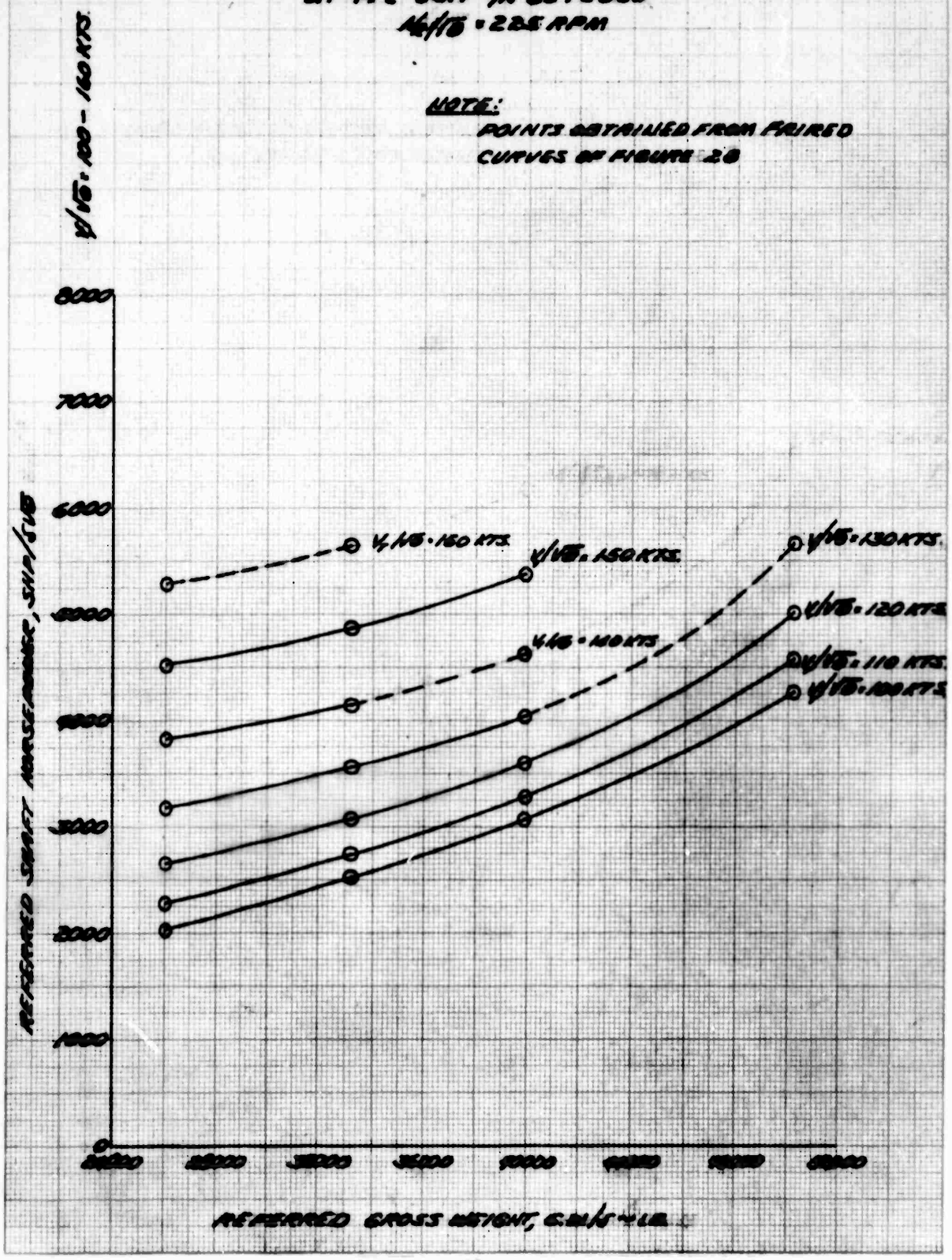


FIGURE 28
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15853
$N_2/\sqrt{\theta}$ = 225 RPM

SYMBOL	AVERAGE REF. GROSS WEIGHT~LB	AVG. PRESSURE ALTITUDE~FT	AVG. OAT ~°C	AVG. C.G. LOCATION~IN/LL	AVG. C_T
○	26060	2310	11.3	331.7 (MIN)	0.003881
□	33250	4480	13.0	330.4 (MID)	0.004951
△	39920	3970	6.7	330.6 (MID)	0.005343
◇	50240	4530	7.7	330.2 (MID)	0.007481

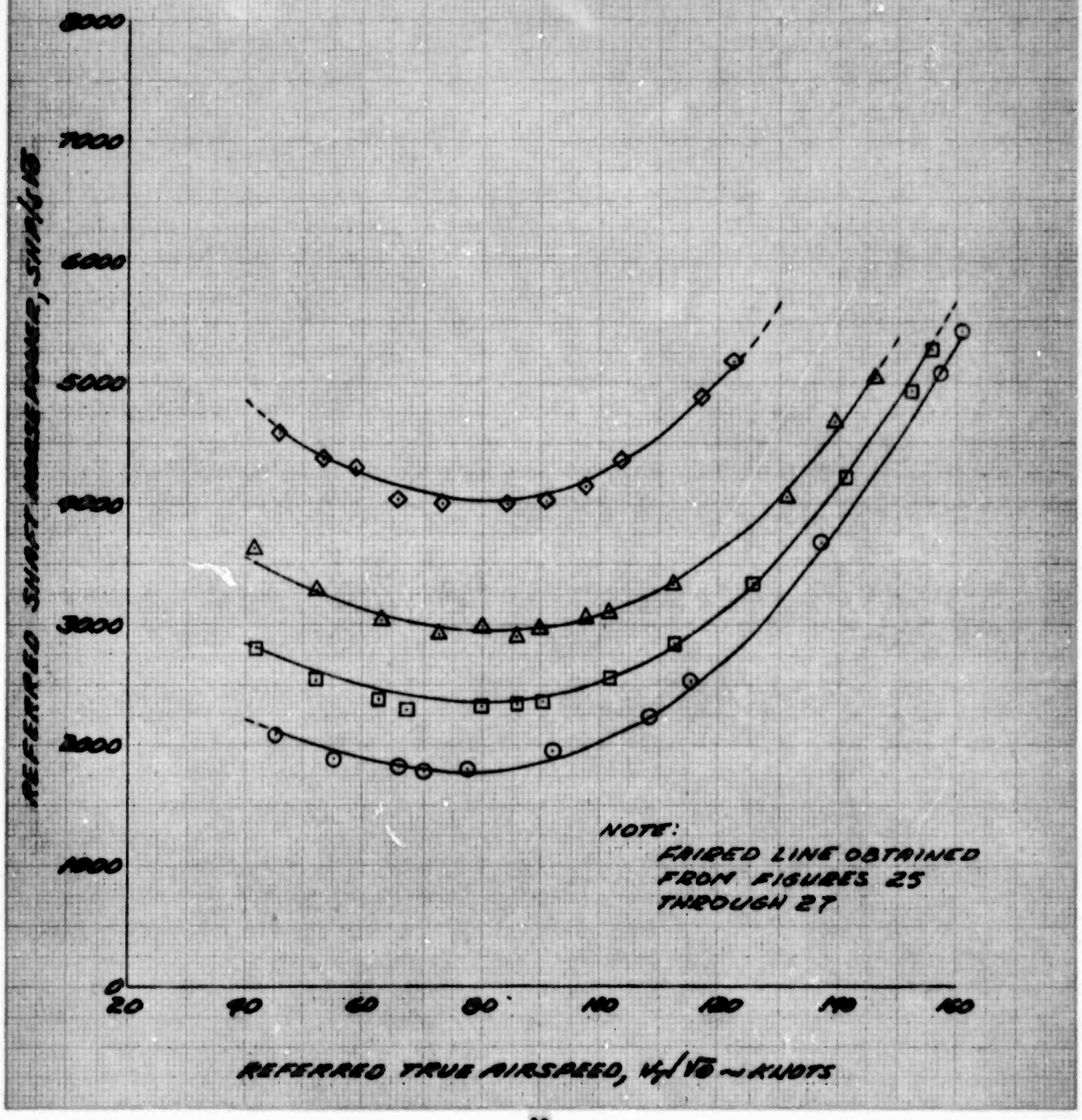


FIGURE 29
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 230 RPM

SYMBOL	ESTIMATED INST. GROSS WEIGHT-LB	AVG. PRESSURE ALTITUDE-FT	AVG. OAT -°C	AVG. C.G. LOCATION-IN.	AVG. C_T
○	33250	5040	2.8	330.6 (AVG)	0.0061738
□	40420	6350	2.7	328.6 (AVG)	0.0085760
△	50260	4510	10.3	330.0 (AVG)	0.0097162

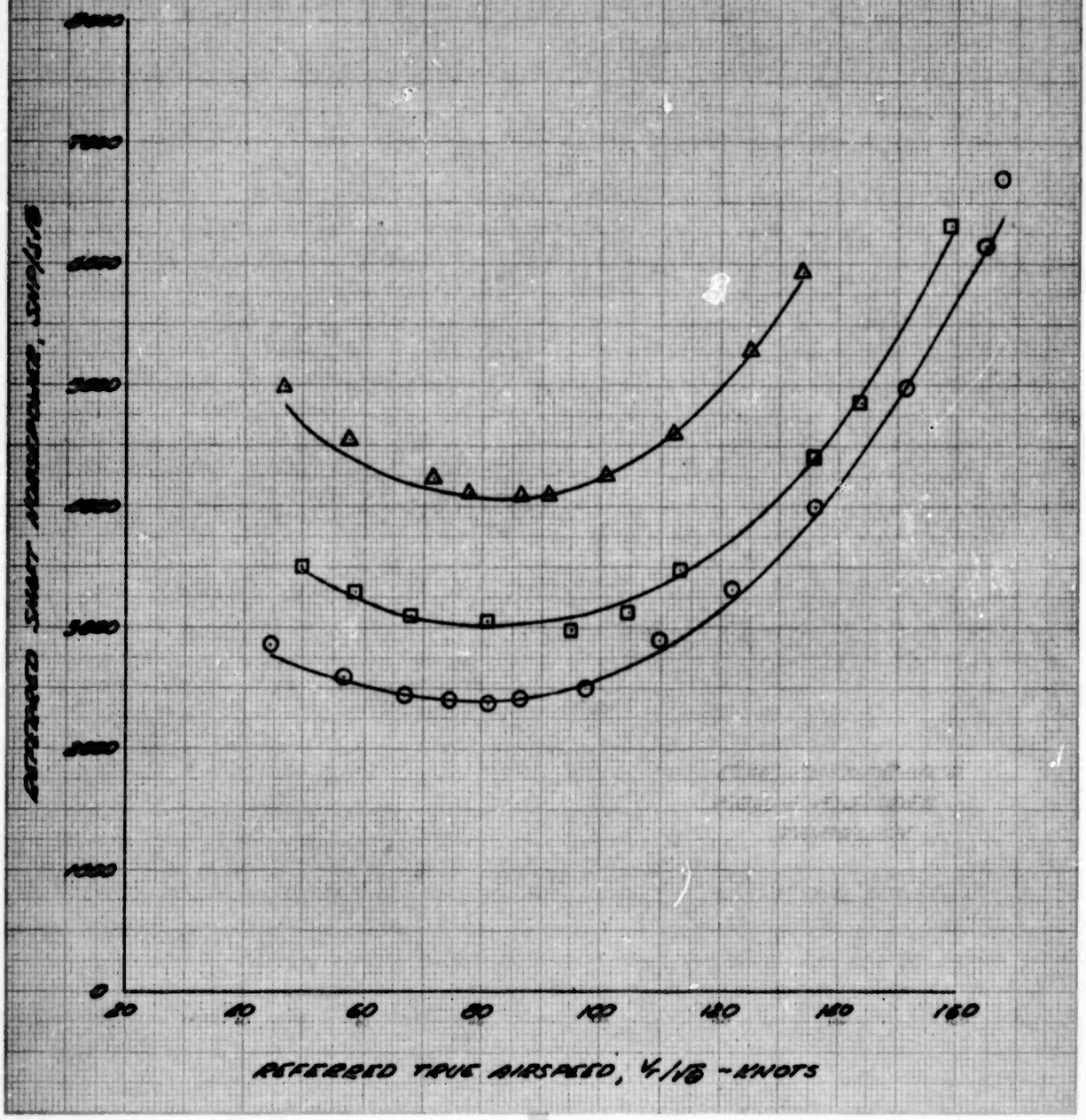


FIGURE 30
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 235 RPM

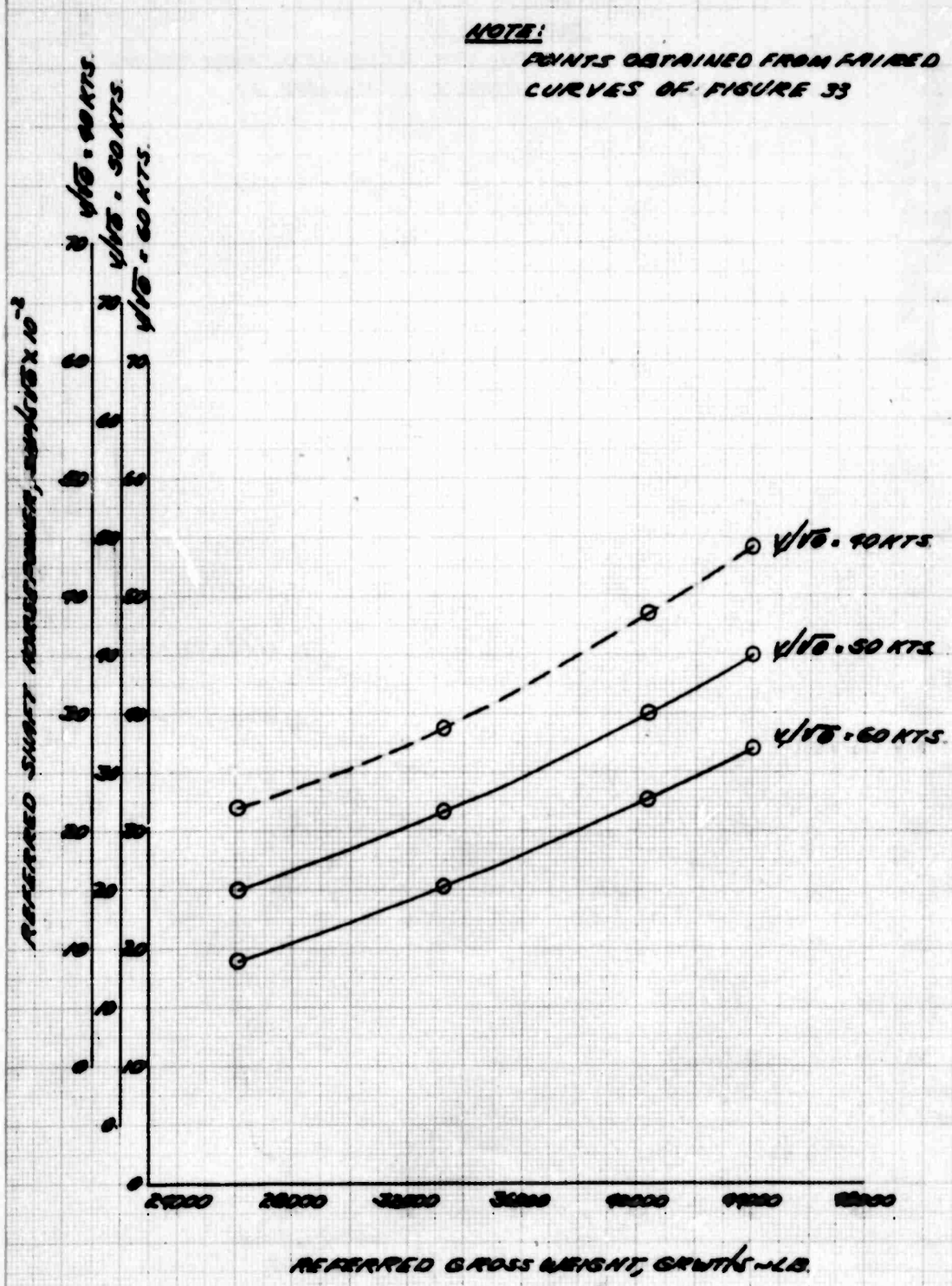


FIGURE 31
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 235 RPM

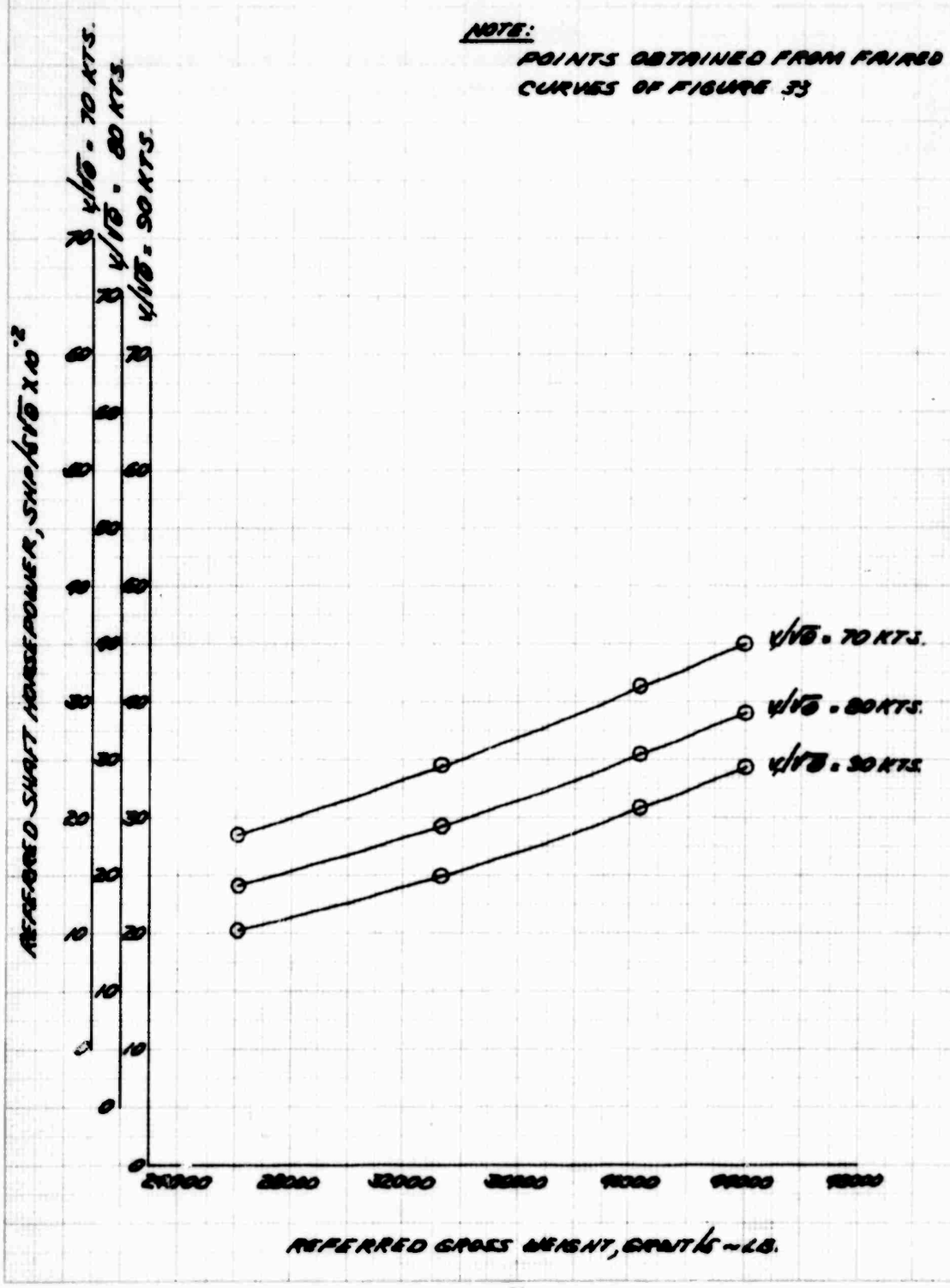


FIGURE 32
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 235 RPM

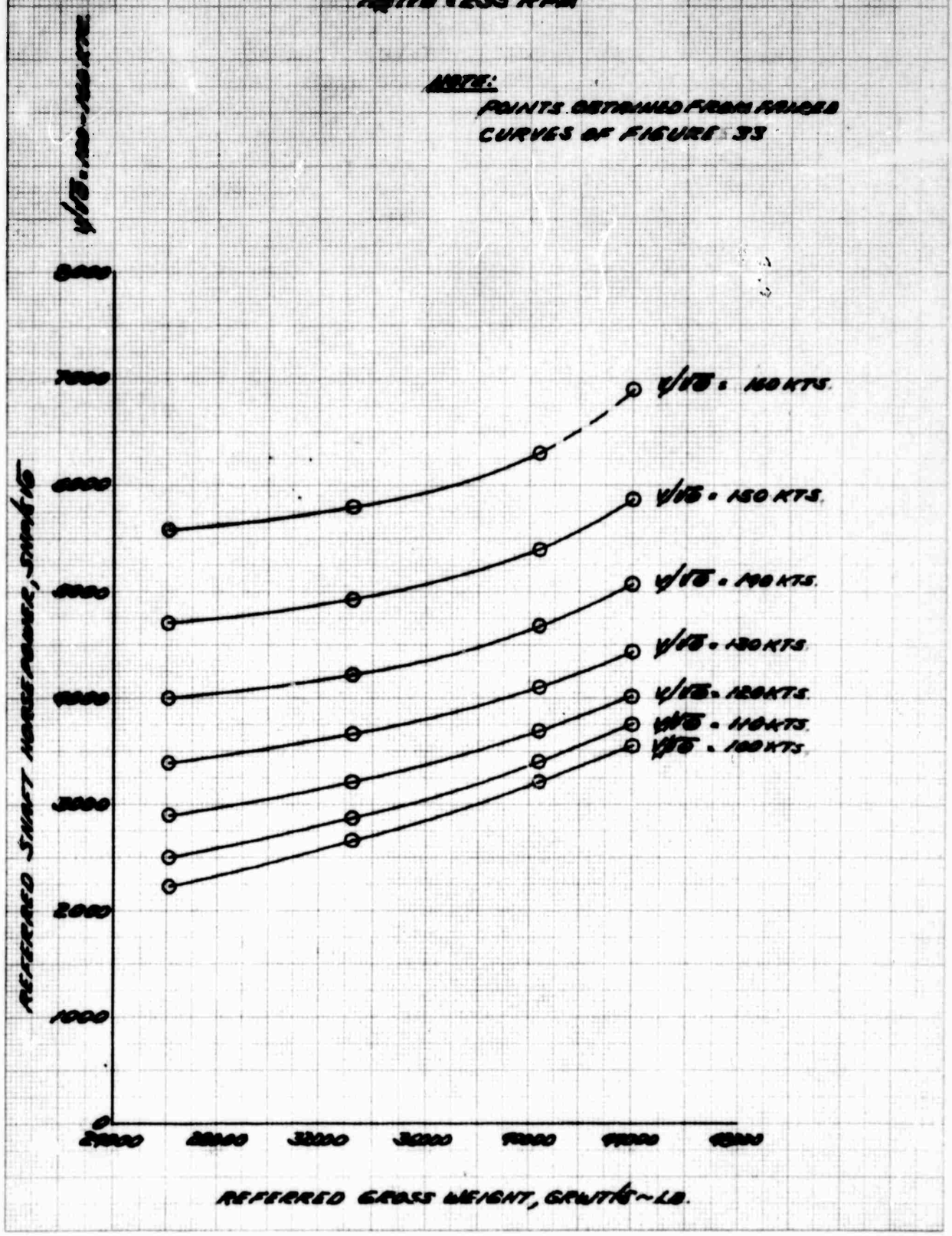


FIGURE 33
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15853
$N_R/\sqrt{\theta}$ = 235 RPM

SYMBOL	AVERAGE AVG. GROSS WEIGHT~LB.	AVG. PRESSURE ALTITUDE~FT.	AVG. OAT ~°C	AVG. C.G. LOCATION~IN.	AVG. C_T
○	26100	2060	20.7	331.7 (MID)	0.003563
□	33280	4730	1.9	329.8 (MID)	0.004542
▲	40410	6390	0.8	331.6 (MID)	0.005515
◇	44080	6670	2.0	329.1 (MID)	0.006012

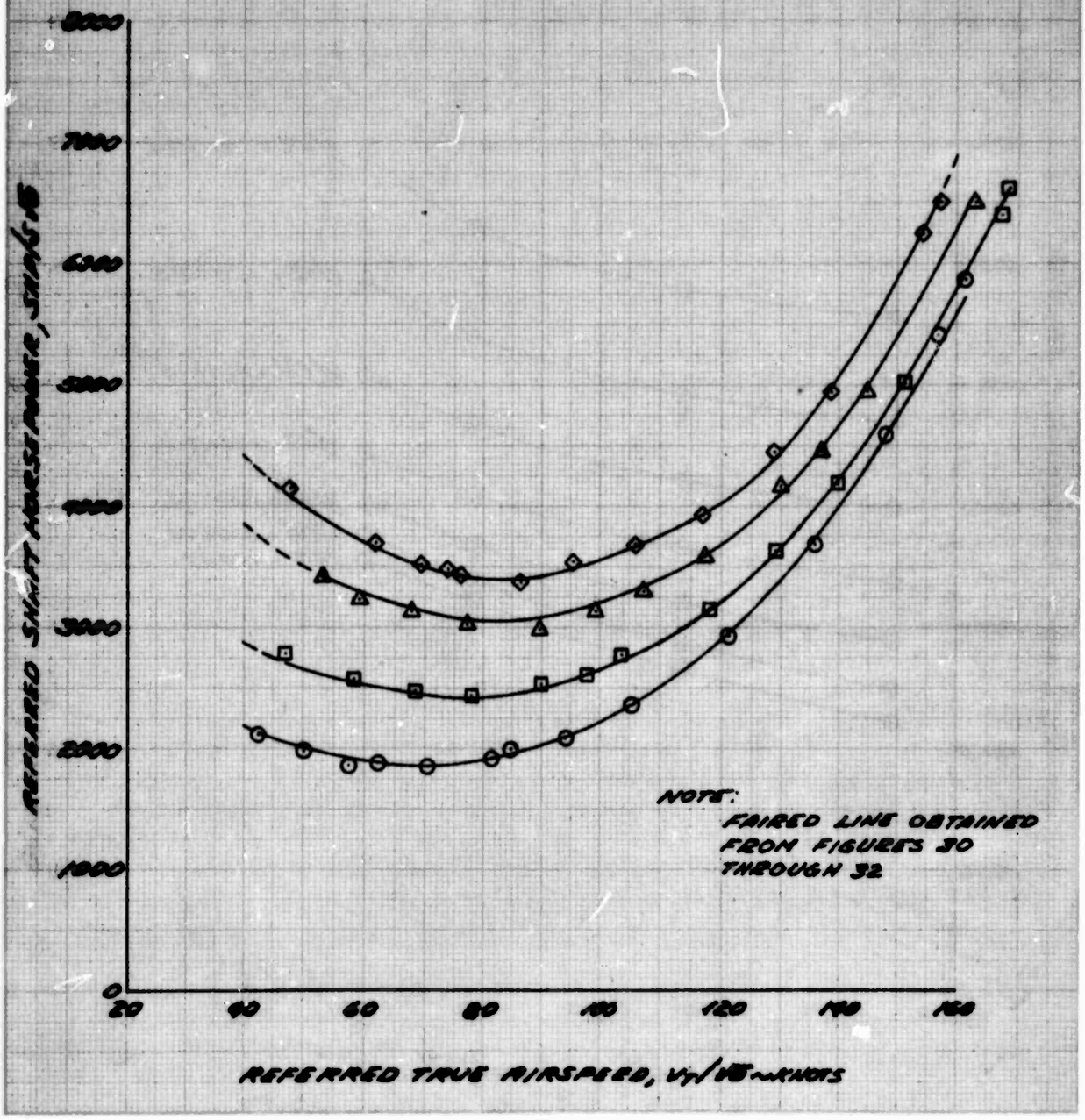


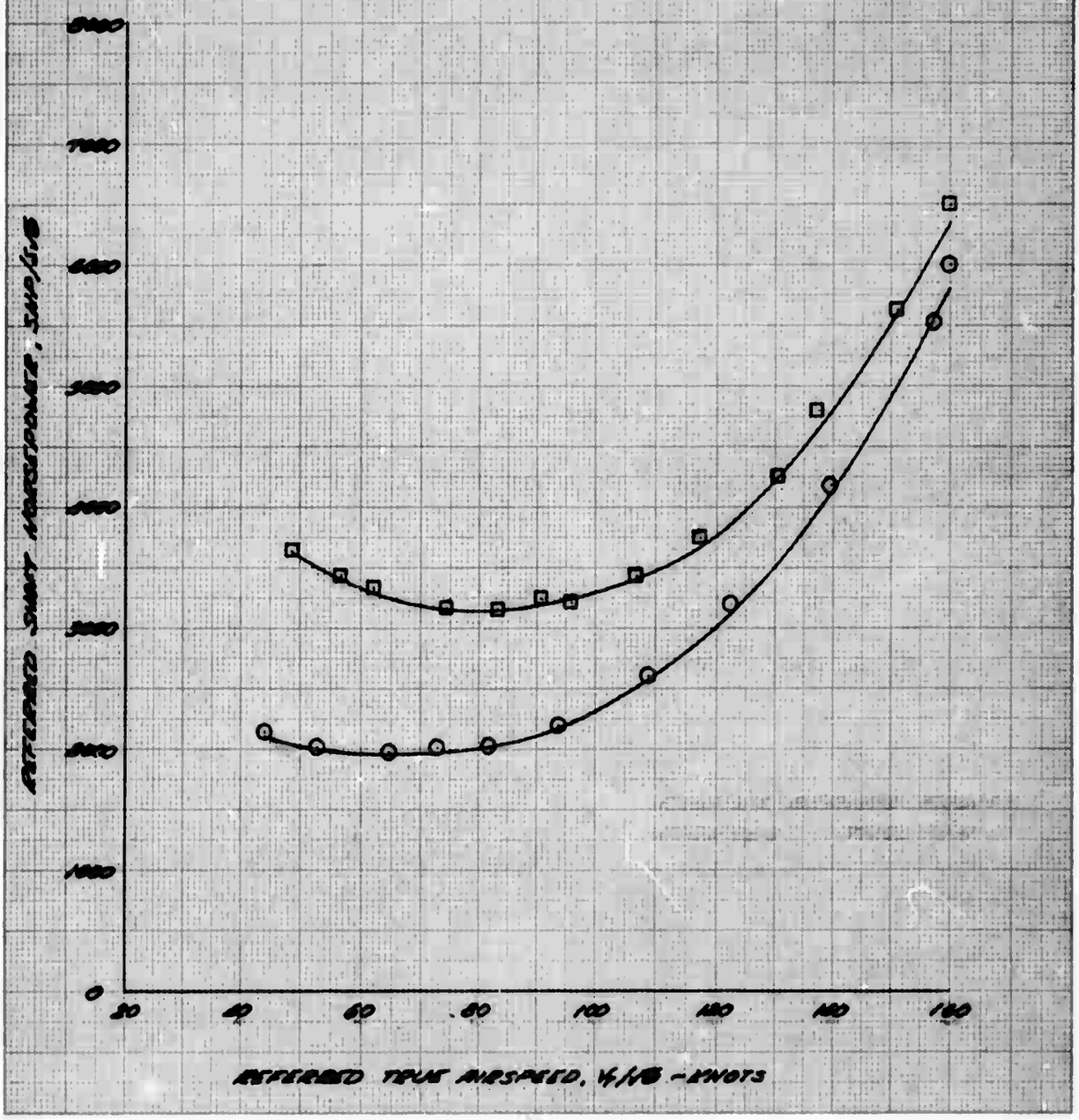
FIGURE 34

REFERRED LEVEL FLIGHT PERFORMANCE

CH-47C USA S/N 68-15839

$N_R/\sqrt{\theta}$ = 240 RPM

SYMBOL	AVERAGE REF. GROSS WEIGHT - LB	AVG. PRESSURE ALTITUDE - FT	AVG. OAT - °C	AVERAGE C.G. LOCATION - IN.	AVG. C_T
○	26040	2140	10.0	331.7 (MID)	0.003608
□	40320	4210	4.5	331.0 (MID)	0.005276



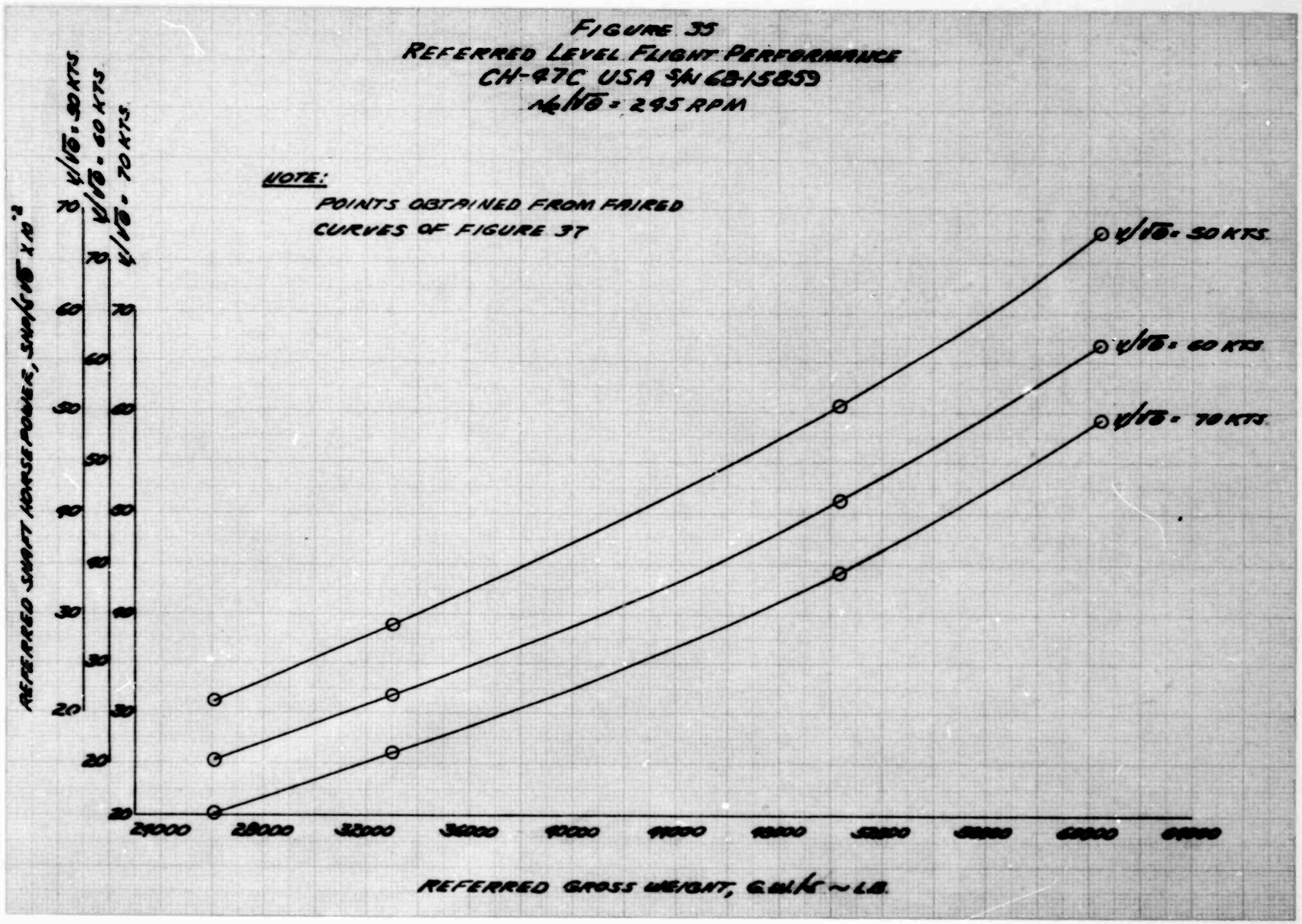


FIGURE 35
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 295 RPM
NOTE:
POINTS OBTAINED FROM FAIRED CURVES OF FIGURE 37
$V/\sqrt{\theta}$ = 50 KTS
$V/\sqrt{\theta}$ = 60 KTS
$V/\sqrt{\theta}$ = 70 KTS
REFERRED SHAFT HORSEPOWER, $SHP/\sqrt{\theta} \times 10^{-2}$
REFERRED GROSS WEIGHT, $G.W./\delta$ ~ LB.
20
30
40
50
60
70
20000
28000
32000
36000
40000
44000
48000
52000
56000
60000
64000

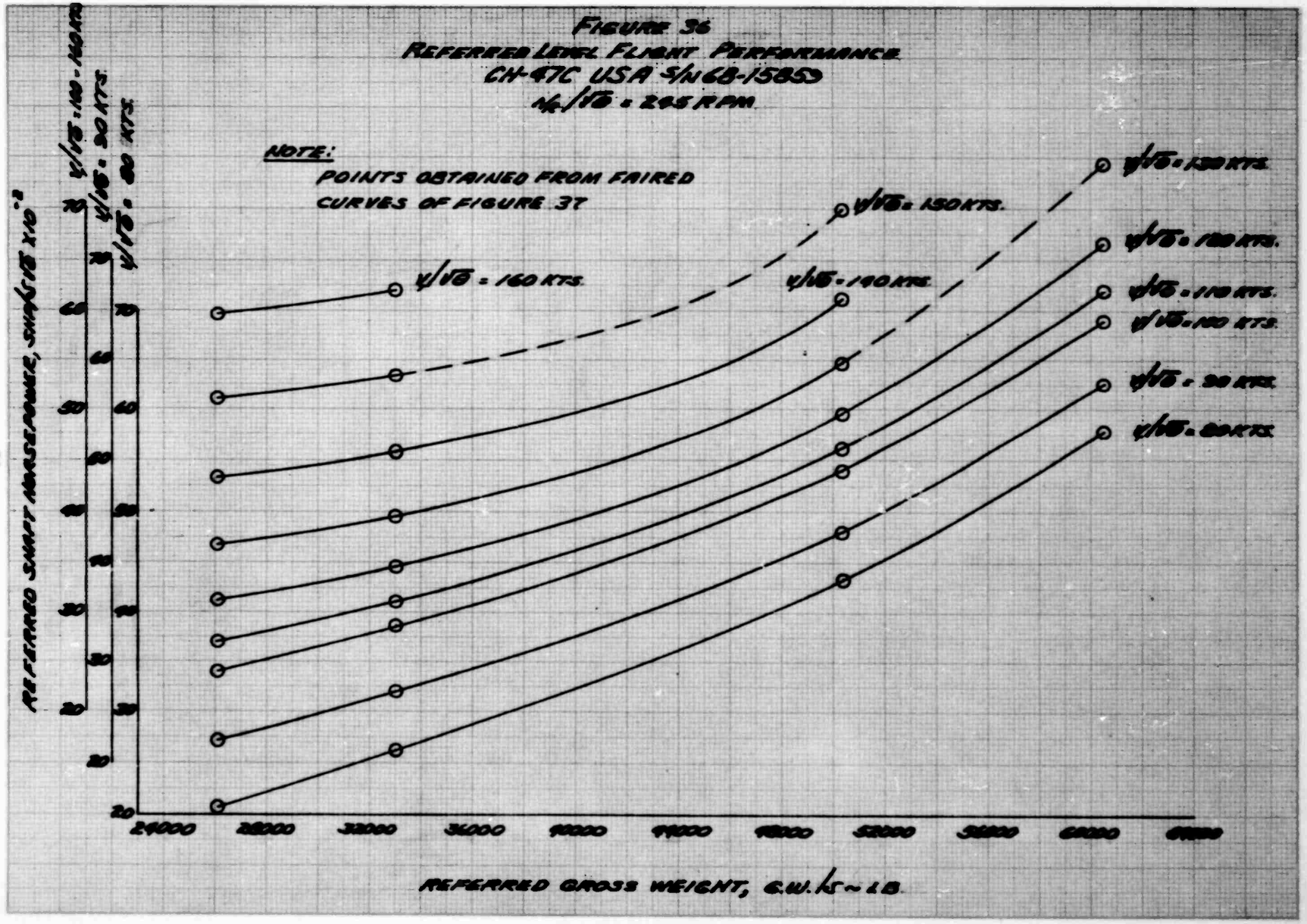


FIGURE 36
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15853
NOTE:
POINTS OBTAINED FROM FAIRED CURVES OF FIGURE 37
V/√θ = 160 KTS.
V/√θ = 150 KTS.
V/√θ = 140 KTS.
V/√θ = 130 KTS.
V/√θ = 120 KTS.
V/√θ = 110 KTS.
V/√θ = 100 KTS.
V/√θ = 90 KTS.
V/√θ = 80 KTS.
V/√θ = 100 - 140 KTS.
V/√θ = 90 KTS.
V/√θ = 80 KTS.
REFERRED SHAFT HORSEPOWER, SHP/δ√θ X10⁻²
REFERRED GROSS WEIGHT, G.W./δ ~ LB.

FIGURE 37
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15853
$N_r/\sqrt{\theta} = 225$ RPM

SYMBOL	AVERAGE GROSS WEIGHT~LB.	AVG. PRESSURE ALTITUDE~FT.	AVG. OAT ~°C	AVG. C.G. LOCATION~IN.	AVG. C_T
○	26120	2190	16.9	331.8(min)	0.003280
□	33060	6900	6.6	330.3(min)	0.004161
△	50360	9670	7.5	330.3(min)	0.006329
◇	60520	8000	6.9	330.3(min)	0.007600

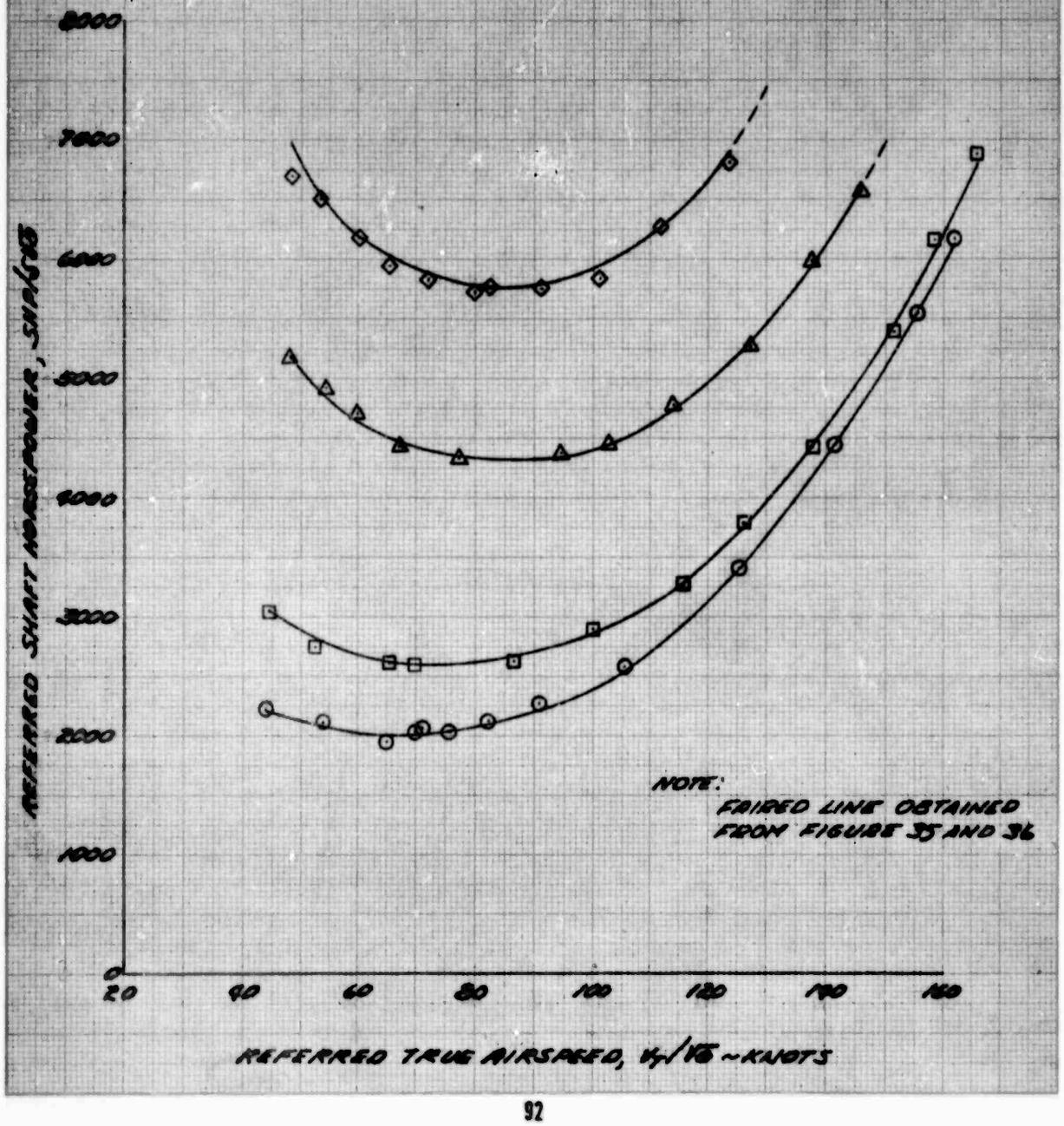
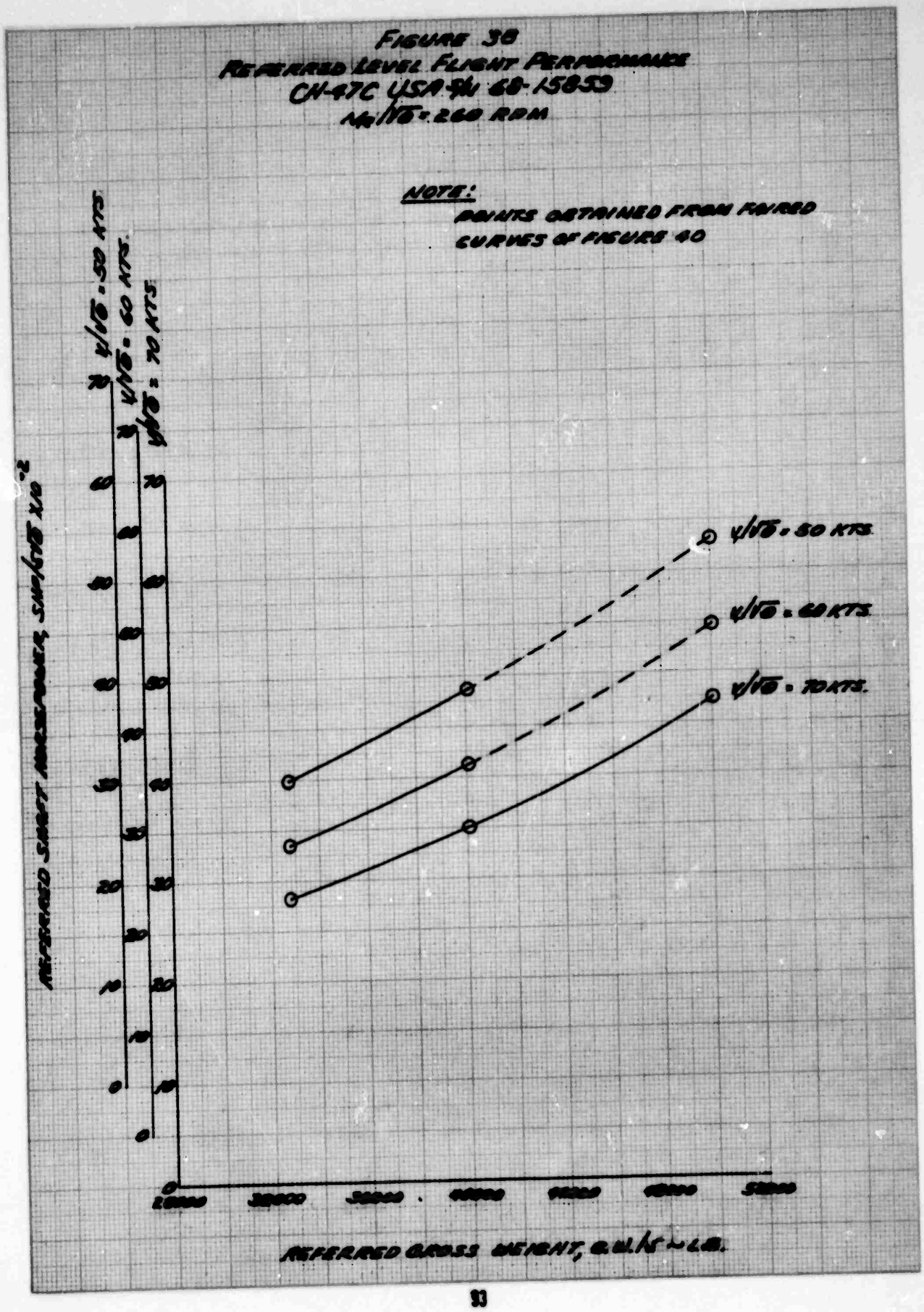


FIGURE 38
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
$N_R/\sqrt{\theta}$ = 260 RPM



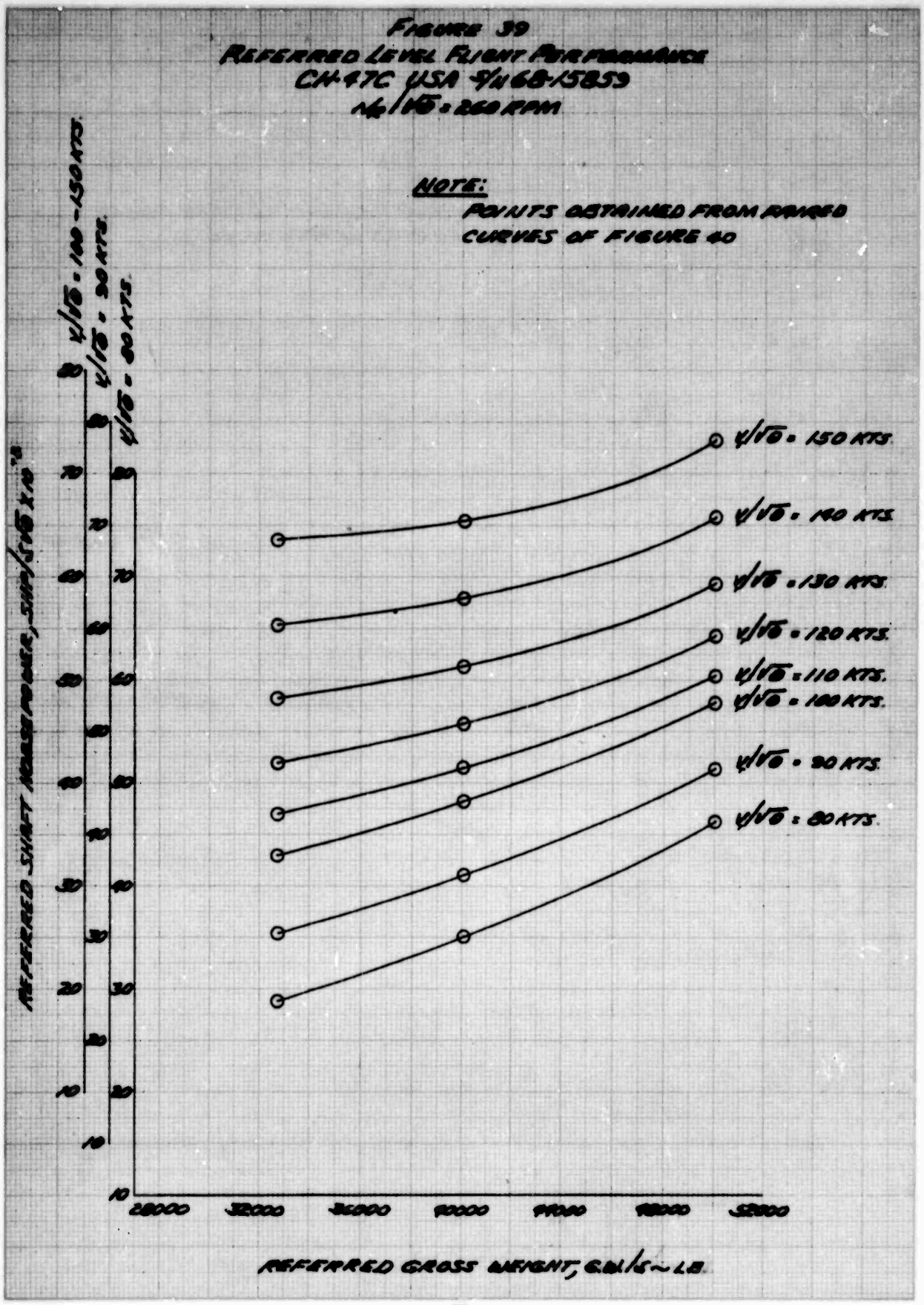


FIGURE 39
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15859
NOTE:
POINTS OBTAINED FROM RAWED CURVES OF FIGURE 40
V/√θ = 100-150 KTS.
V/√θ = 90 KTS.
V/√θ = 80 KTS.
V/√θ = 150 KTS.
V/√θ = 140 KTS.
V/√θ = 130 KTS.
V/√θ = 120 KTS.
V/√θ = 110 KTS.
V/√θ = 100 KTS.
V/√θ = 90 KTS.
V/√θ = 80 KTS.
REFERRED SHAFT HORSEPOWER, SHP/δ√θ x 10^-3
REFERRED GROSS WEIGHT, G.W./δ ~ LB.

FIGURE 40
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15839
$N_R/\sqrt{\theta}$ = 260 RPM

SYMBOL	AVERAGE GROSS WEIGHT~LB	AVG. PRESSURE ALTITUDE~FT	AVG. OAT ~°C	AVG. C.G. LOCATION~IN.	AVG. C_T
○	32760	4400	-20.2	331.0(FWD)	0.003653
□	40080	3380	-19.1	331.0(FWD)	0.004969
△	50030	7690	-18.1	330.9(FWD)	0.005579

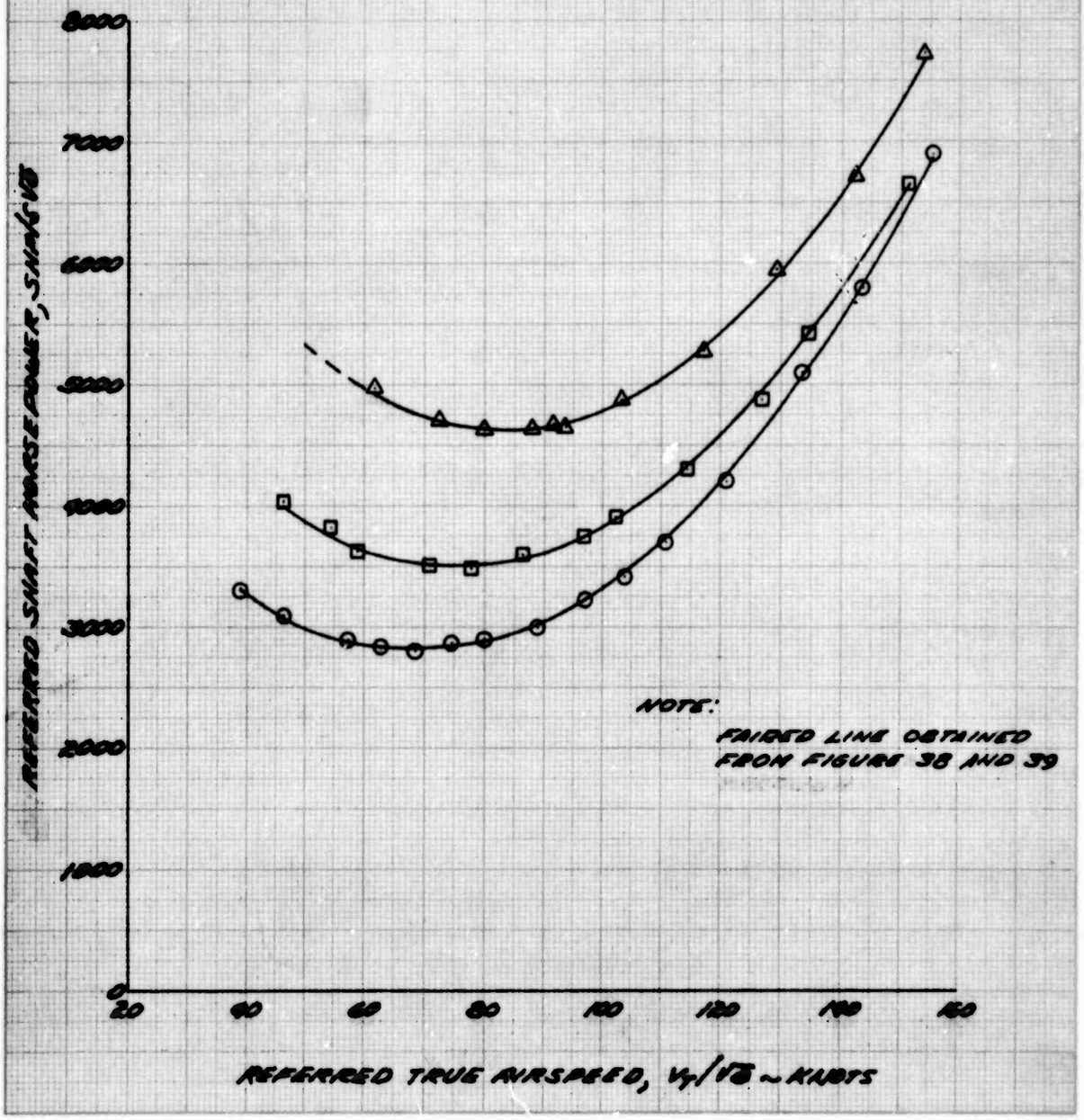


FIGURE 41
REFERRED LEVEL FLIGHT PERFORMANCE
CH-47C USA S/N 68-15853
$N_R/\sqrt{\theta}$ = 268 RPM

SYMBOL	AVERAGE ACT. GROSS WEIGHT~LB	AVG. PRESSURE ALTITUDE~FT.	AVG. OAT ~°C	AVG. C.G. LOCATION~IN.	AVG. C_T
○	48010	4140	-20.5	330.5(FWD)	0.009133
□	50110	7750	-18.9	329.5(FWD)	0.005260

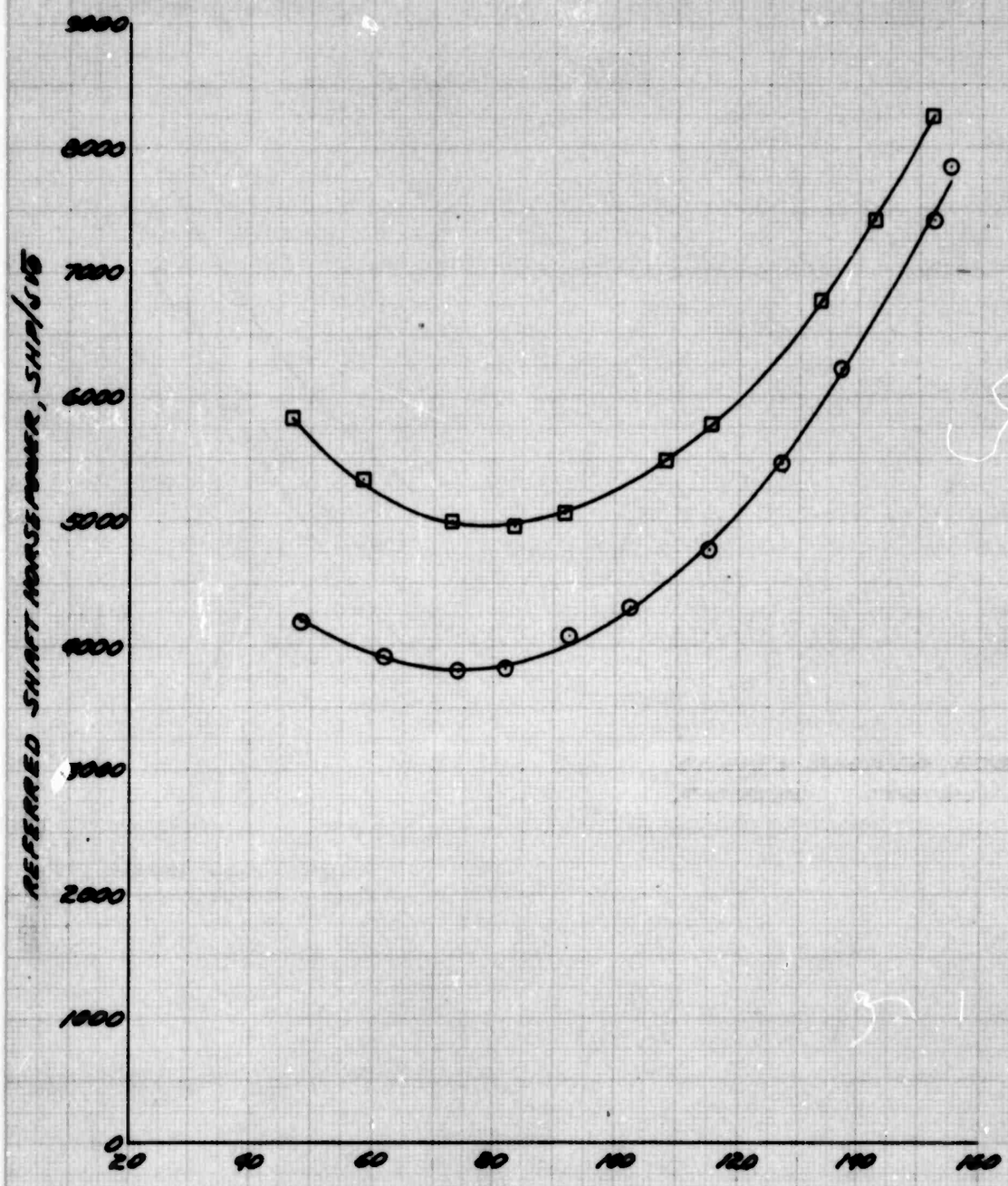
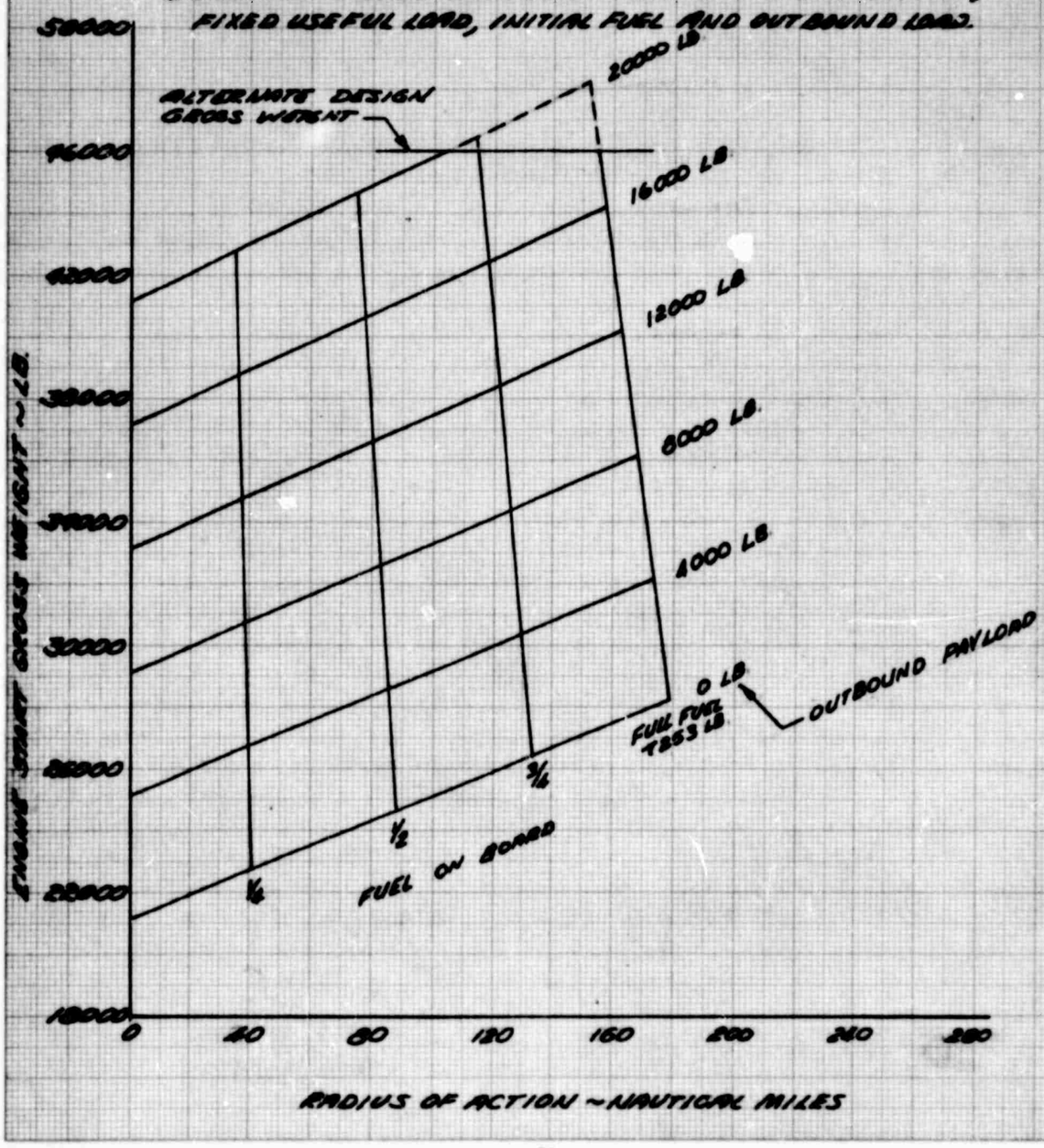


FIGURE 42
RANGE SUMMARY
CH-47C USA S/N 68-15853
SEA LEVEL STANDARD DAY
ROTOR SPEED = 245 RPM
MID CG

NOTES:

1. BASED ON FIGURES 44 AND 58
2. MISSION PROFILE: CRUISE AIRSPEED AT 0.33 HAMPP$_{MAX}$, 2-MINUTE WARM UP AT NORMAL RATED POWER, INBOUND LOAD = 1/2 OUTBOUND LOAD, RETURN WITH 10% INITIAL FUEL REMAINING.
3. ENG. START GROSS WEIGHT CONSISTS OF EMPTY WEIGHT, FIXED USEFUL LOAD, INITIAL FUEL AND OUTBOUND LOAD.



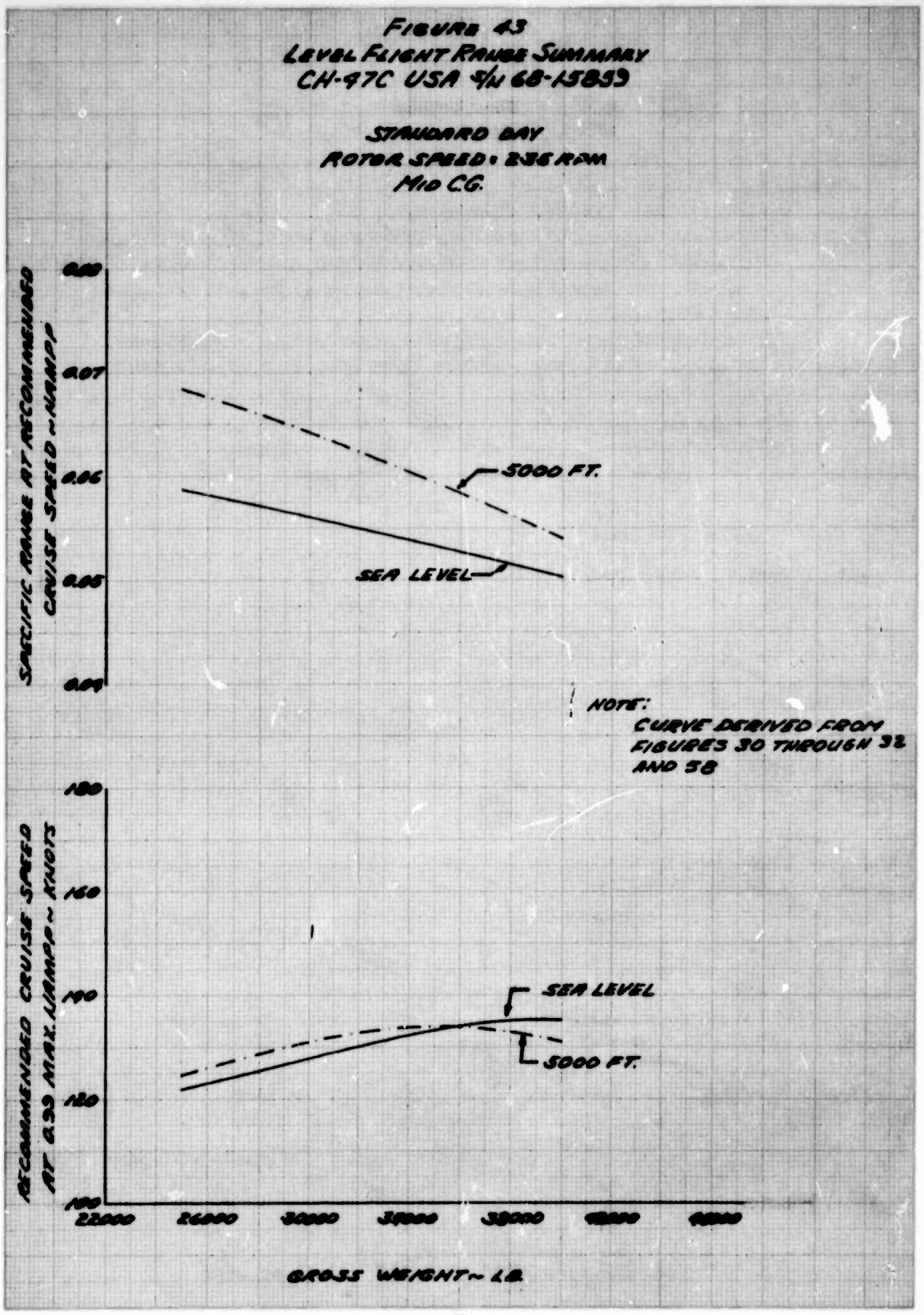


FIGURE 43
LEVEL FLIGHT RANGE SUMMARY
CH-47C USA S/N 68-15839
STANDARD DAY
ROTOR SPEED = 235 RPM
MID C.G.
SPECIFIC RANGE AT RECOMMENDED CRUISE SPEED ~ NAM/LB
0.07
0.06
0.05
5000 FT.
SEA LEVEL
NOTE:
CURVE DERIVED FROM FIGURES 30 THROUGH 32 AND 58
RECOMMENDED CRUISE SPEED AT 0.99 MAX. NAMPP ~ KNOTS
160
140
120
SEA LEVEL
5000 FT.
26000
30000
38000
GROSS WEIGHT ~ LB.

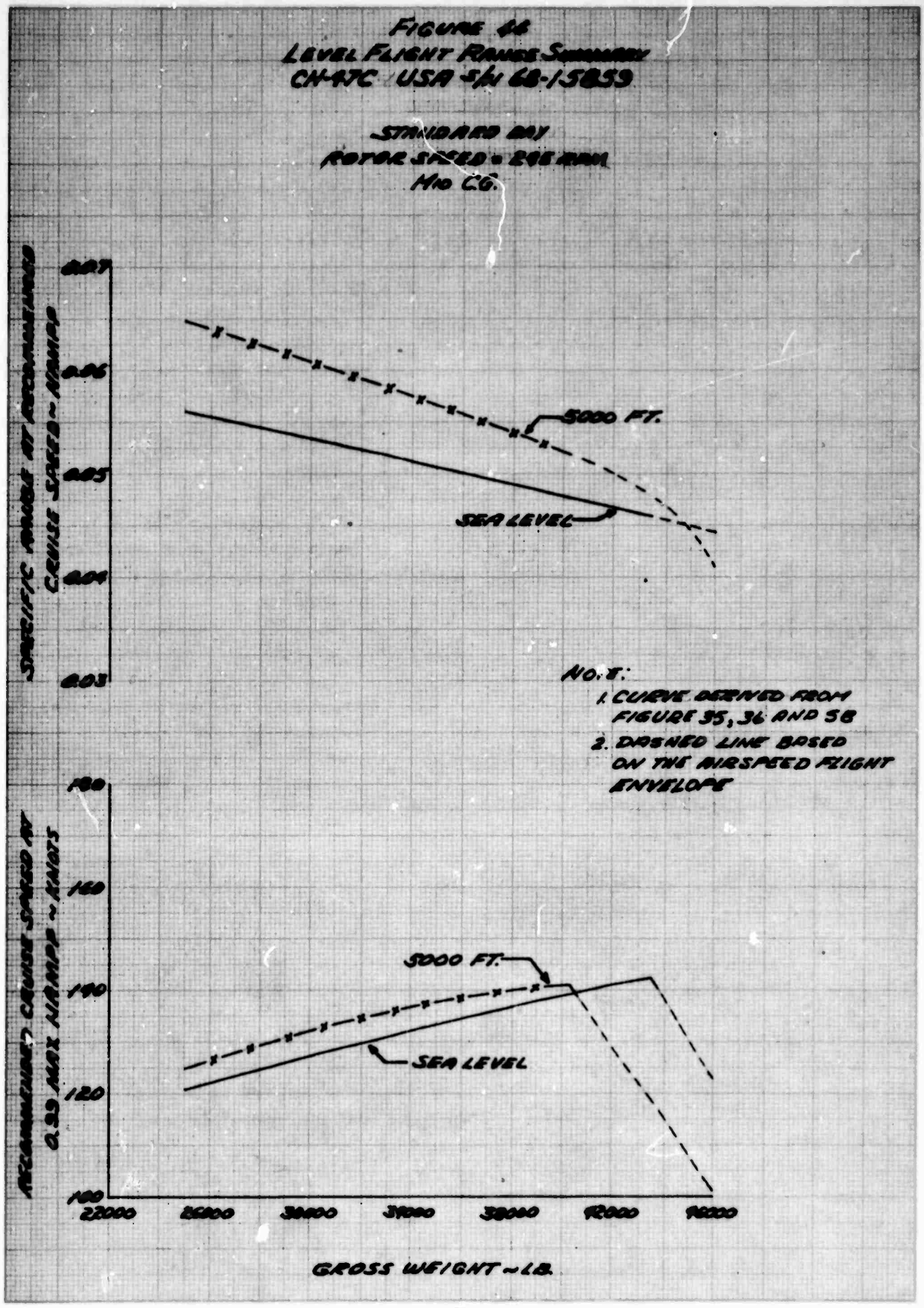


FIGURE 44
LEVEL FLIGHT RANGE SUMMARY
CH-47C USA S/N 68-15859
STANDARD DAY
ROTOR SPEED = 225 RPM
MID C.G.
SPECIFIC RANGE AT RECOMMENDED CRUISE SPEED ~ NM/LB
0.07
0.06
0.05
0.04
0.03
5000 FT.
SEA LEVEL
NOTE:
1. CURVE DERIVED FROM FIGURE 35, 36 AND 58
2. DASHED LINE BASED ON THE AIRSPEED FLIGHT ENVELOPE
RECOMMENDED CRUISE SPEED AT 0.39 MAX HANDLE ~ KNOTS
180
160
140
120
100
5000 FT.
SEA LEVEL
22000
26000
30000
34000
38000
42000
46000
GROSS WEIGHT ~ LB.

FIGURE 45
AUTOROTATIONAL DESCENT PERFORMANCE
CH-47C USA S/N 68-15859
MID CG

NOTE: MAXIMUM CONTINUOUS ROTOR SPEED LIMIT = 261 RPM

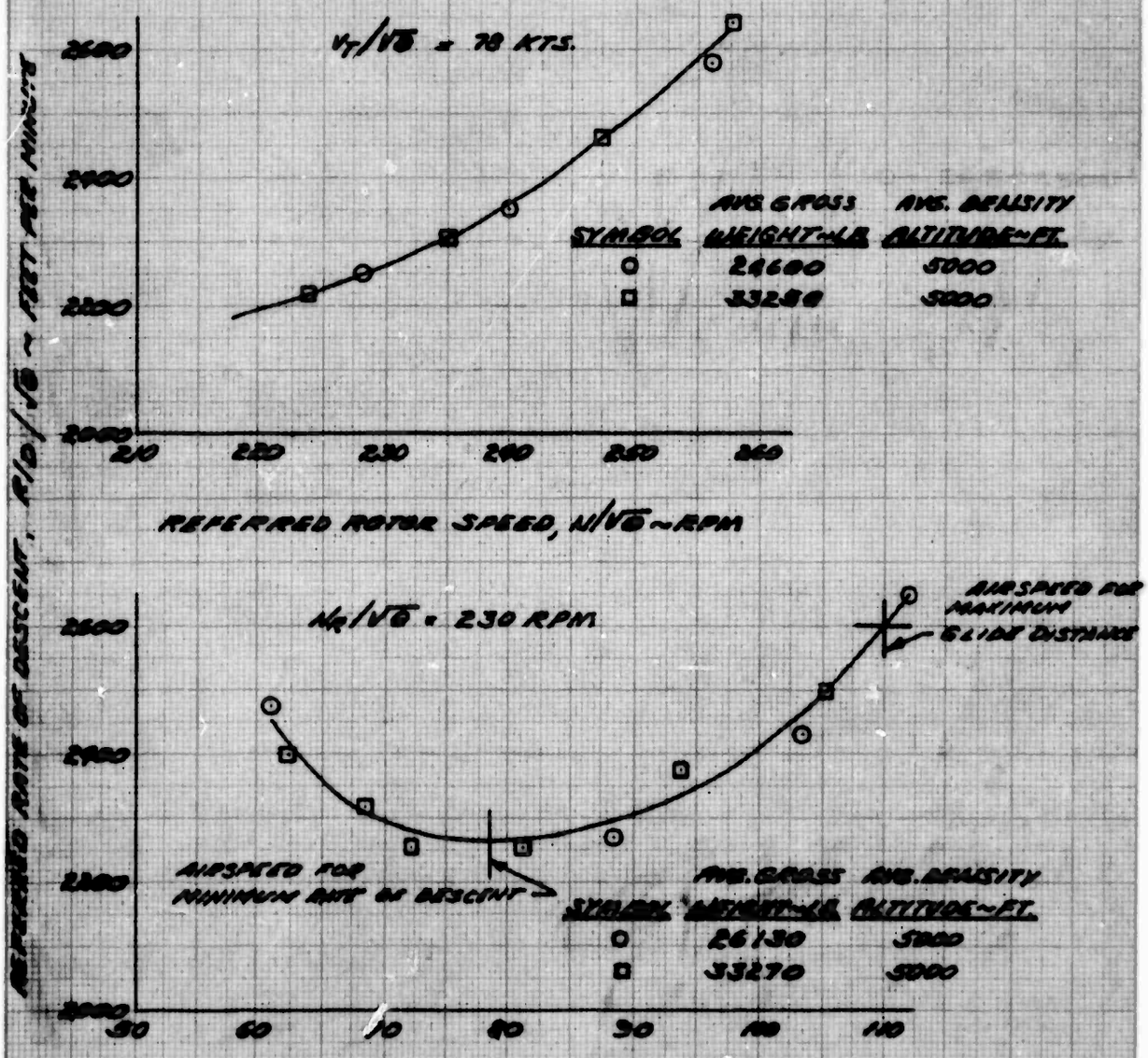


FIGURE 46
NORMAL RATED SHAFT HORSEPOWER AVAILABLE
CH-47C USA S/N 68-15859
LYCOMING T53-L-11 ENGINE
ROTOR SPEED 225 RPM

NOTES:
1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO. 129.27A T53-L-11 SHAFT TURBINE ENGINE.

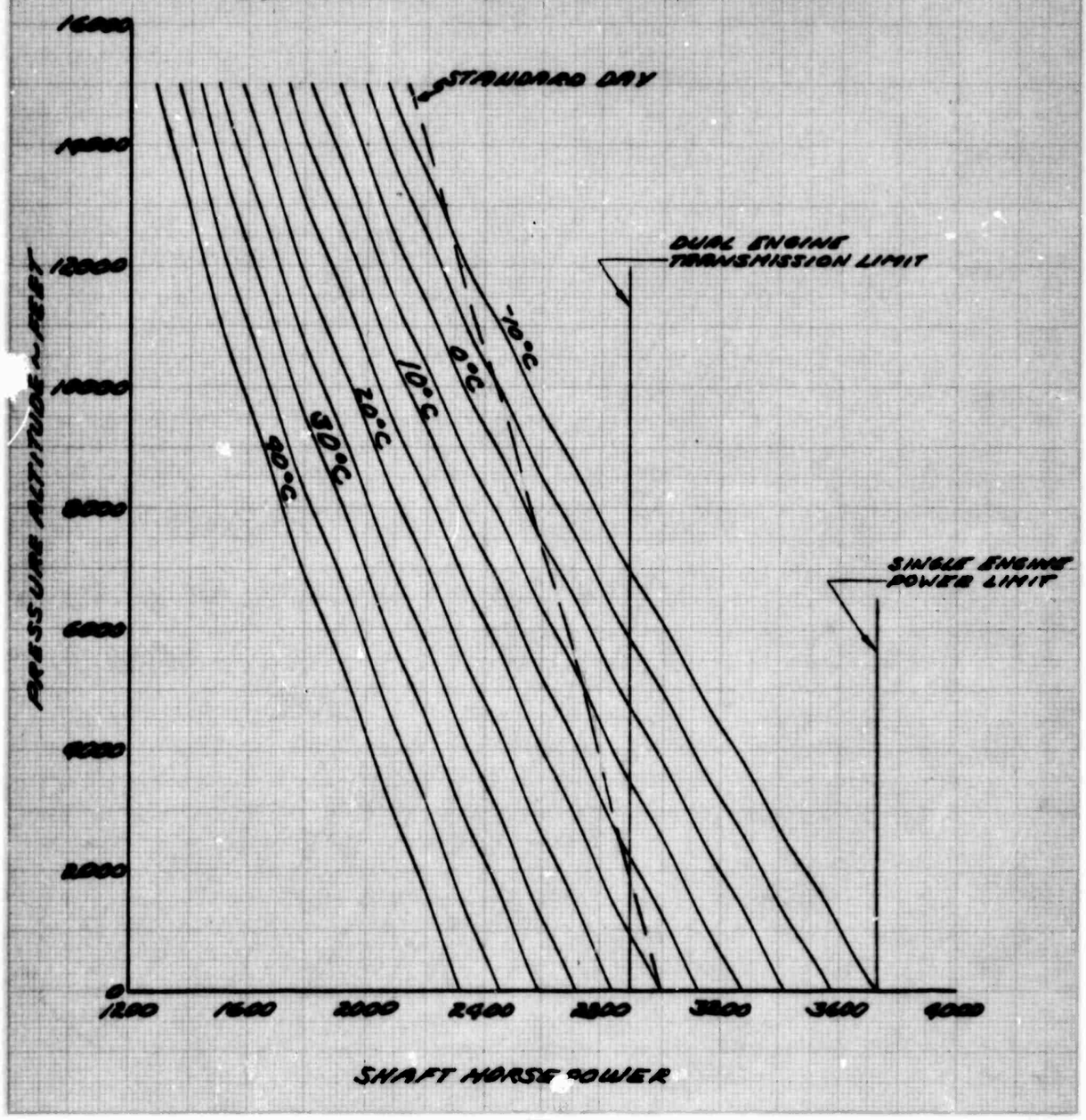


FIGURE 47
MILITARY RATED SHAFT HORSEPOWER AVAILABLE
CH-47C USA S/N 68-15853
LYCOMING T53-L-11 ENGINE
ROTOR SPEED 225 RPM

NOTES:

1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES.
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO 124.27A. T55-L-11 SHAFT TURBINE ENGINE.

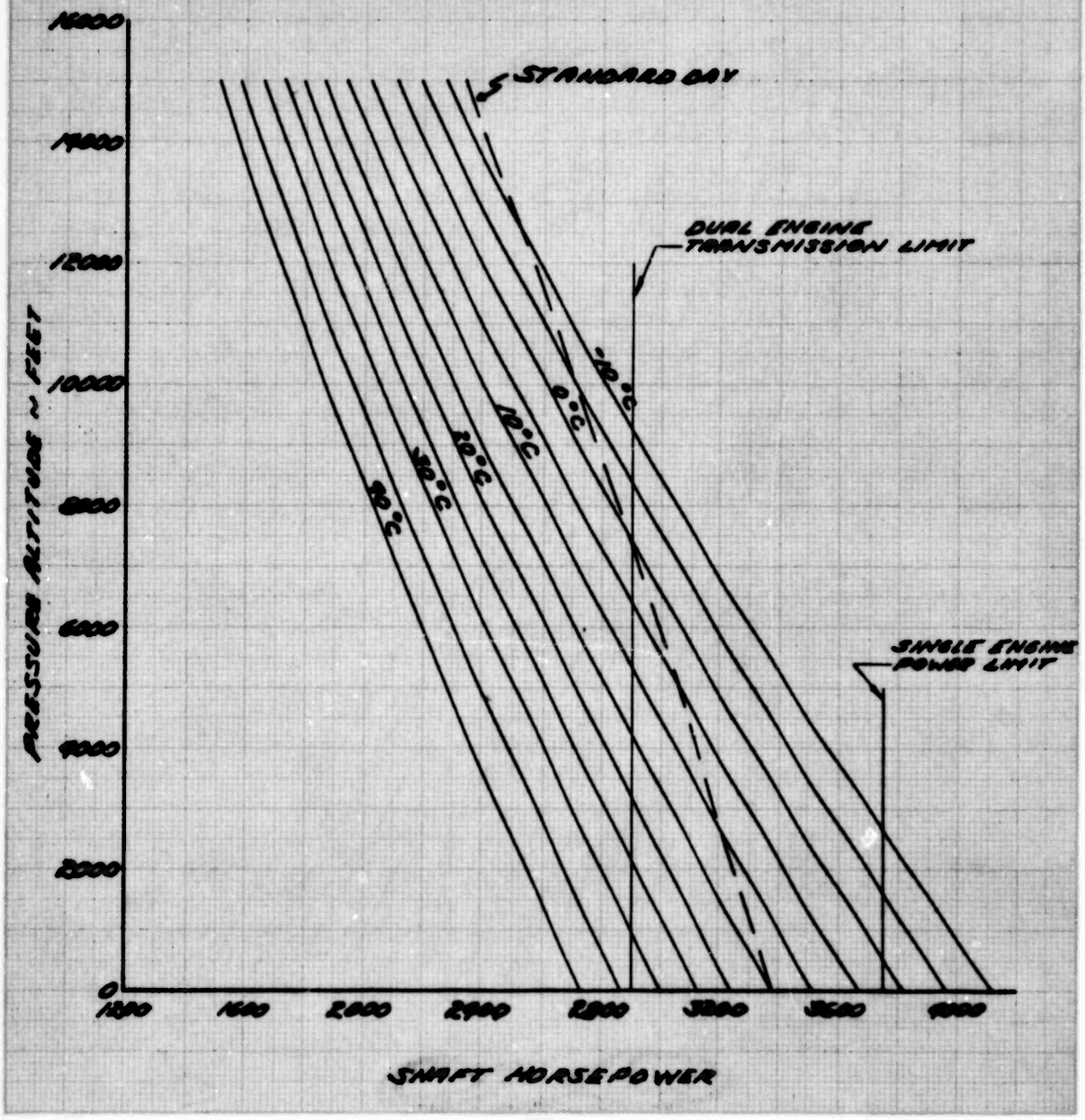


FIGURE 49
MAXIMUM RATED SHAFT HORSEPOWER AVAILABLE
CH-47C USA S/N 68-15859
LYCOMING T53-L-11 ENGINE
ROTOR SPEED = 235 RPM

NOTES:
1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES.
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO. 129.2 TA. T53-L-11 SHAFT TURBINE ENGINE.

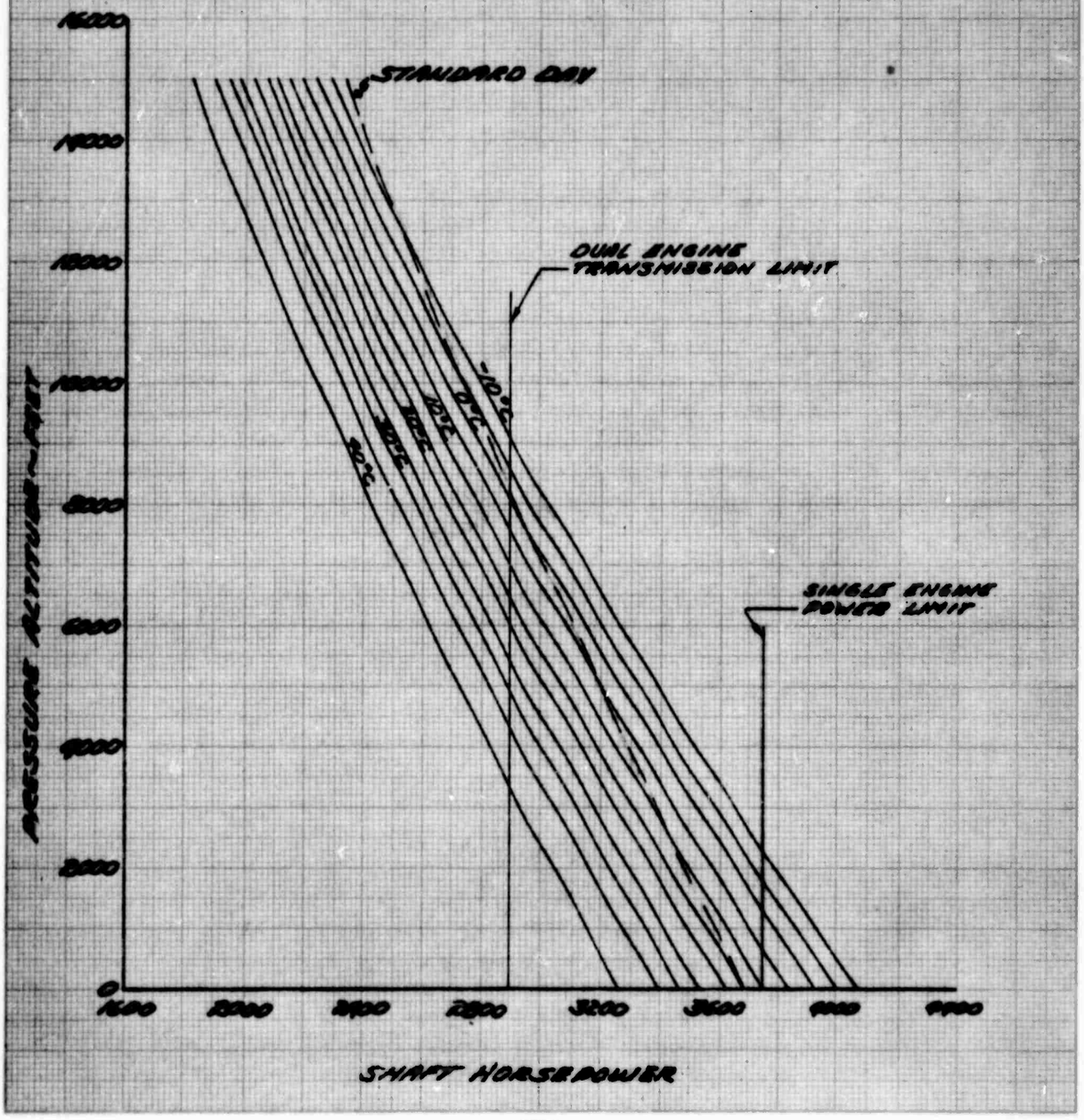


FIGURE 49

MAXIMUM RATED SHAFT HORSEPOWER AVAILABLE

CH-47C USA S/N 68-15853

LYCOMING T55-L-11 ENGINE

ROTOR SPEED = 225 RPM

NOTES:

1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES.
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO. 129.27A. T55-L-11 SHAFT TURBINE ENGINE.

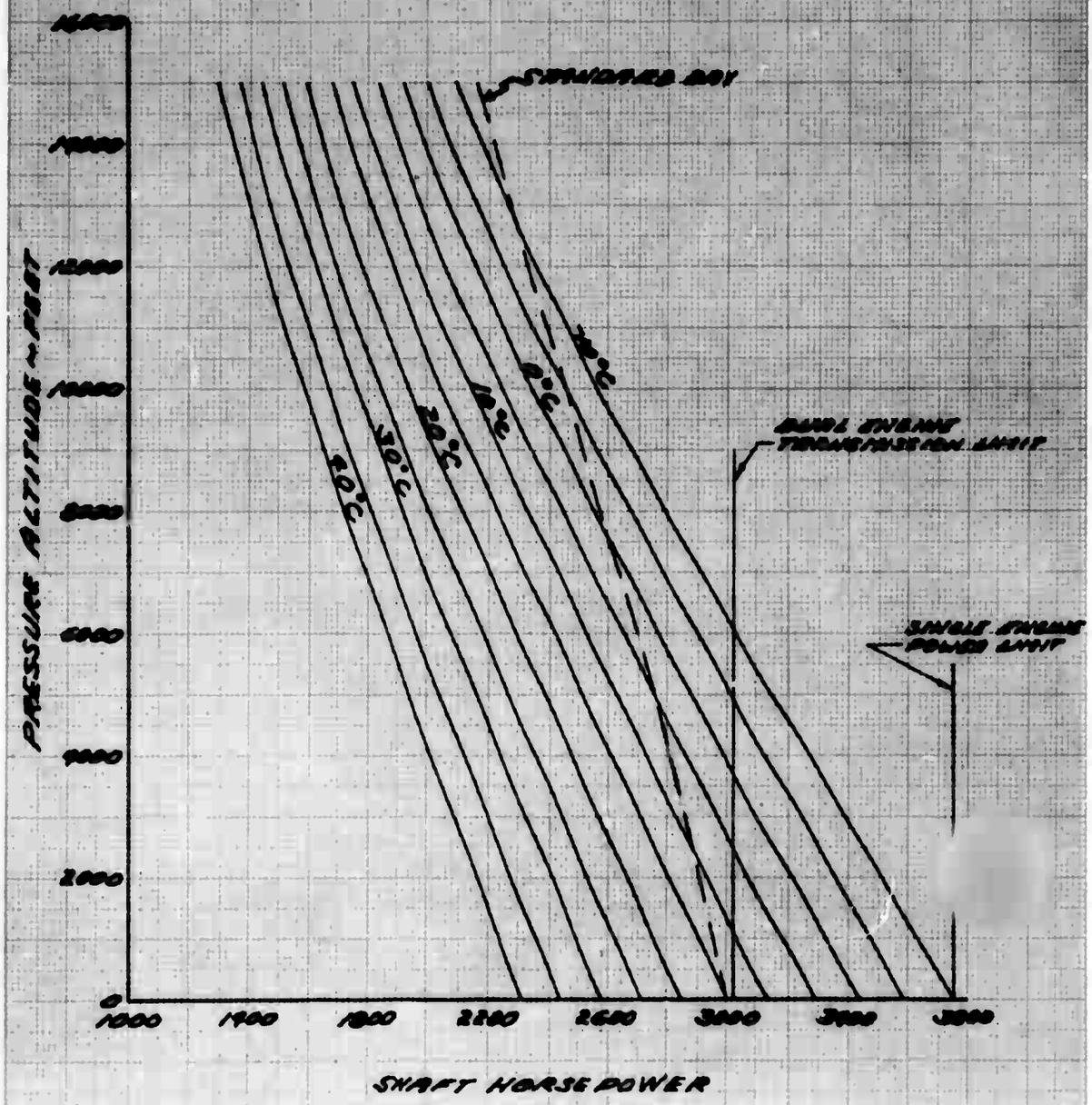


FIGURE 50
MILITARY RATED SHAFT HORSEPOWER AVAILABLE
CH-47C USA S/N 68-15853
LYCOMING T55-L-11 ENGINE
ROTOR SPEED = 245 RPM

NOTES:

1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES.
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO. 126.2 PA. T55-L-11 SHAFT TURBINE ENGINE.

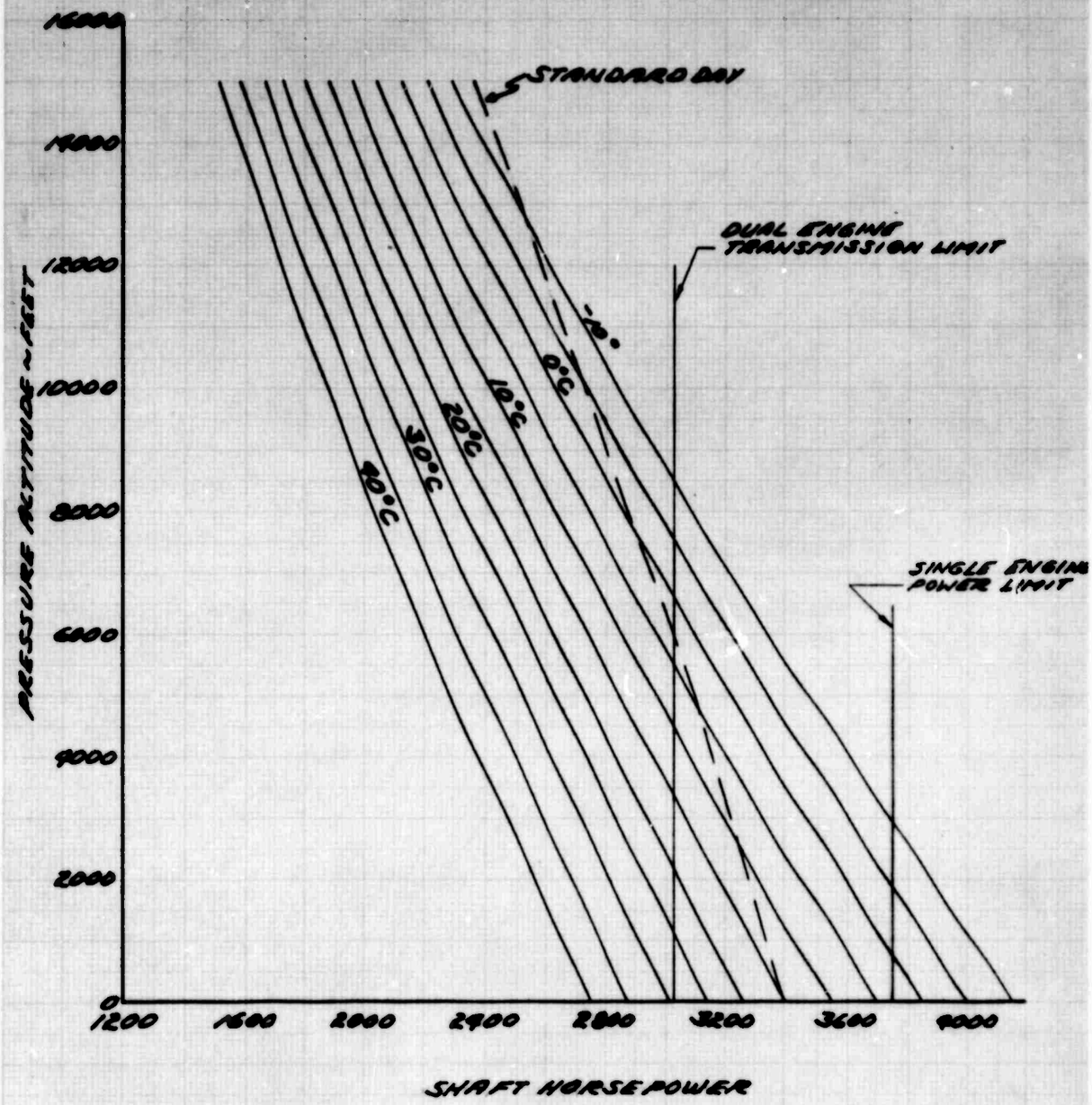


FIGURE 31

MAXIMUM BOARD SHAFT HORSEPOWER AVAILABLE

CH-47C USA S/N 68-15869

LYCOMING T55-L-11 ENGINE

ROTOR SPEED 245 RPM

NOTES:

1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES.
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO. 12427A. T55-L-11 SHAFT TURBINE ENGINE.

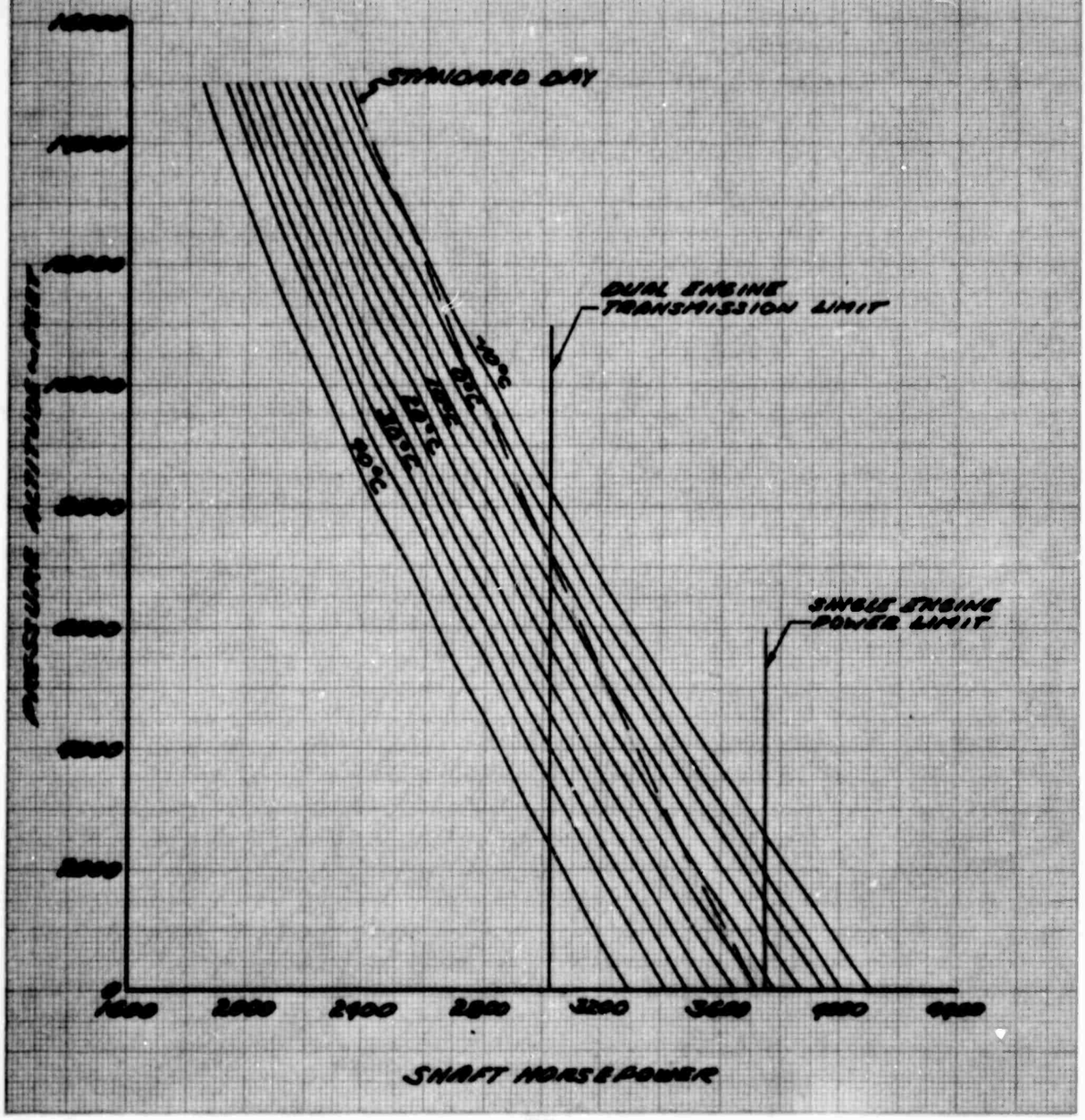


FIGURE 58
SHAFT HORSEPOWER AVAILABLE WITH RAM EFFECTS
CH-47C USA S/N 68-15859
LYCOMING T55-L-11 ENGINE
NORMAL RATED POWER
ROTOR SPEED = 235 RPM

NOTES:
1. STANDARD DAY.
2. ENGINE INLET LOSSES BASED ON FIGURE 59 AND 60
3. BASED ON AVCO LYCOMING MODEL SPEC. NO 129.27A. T55-L-11 SHAFT TURBINE ENGINE.

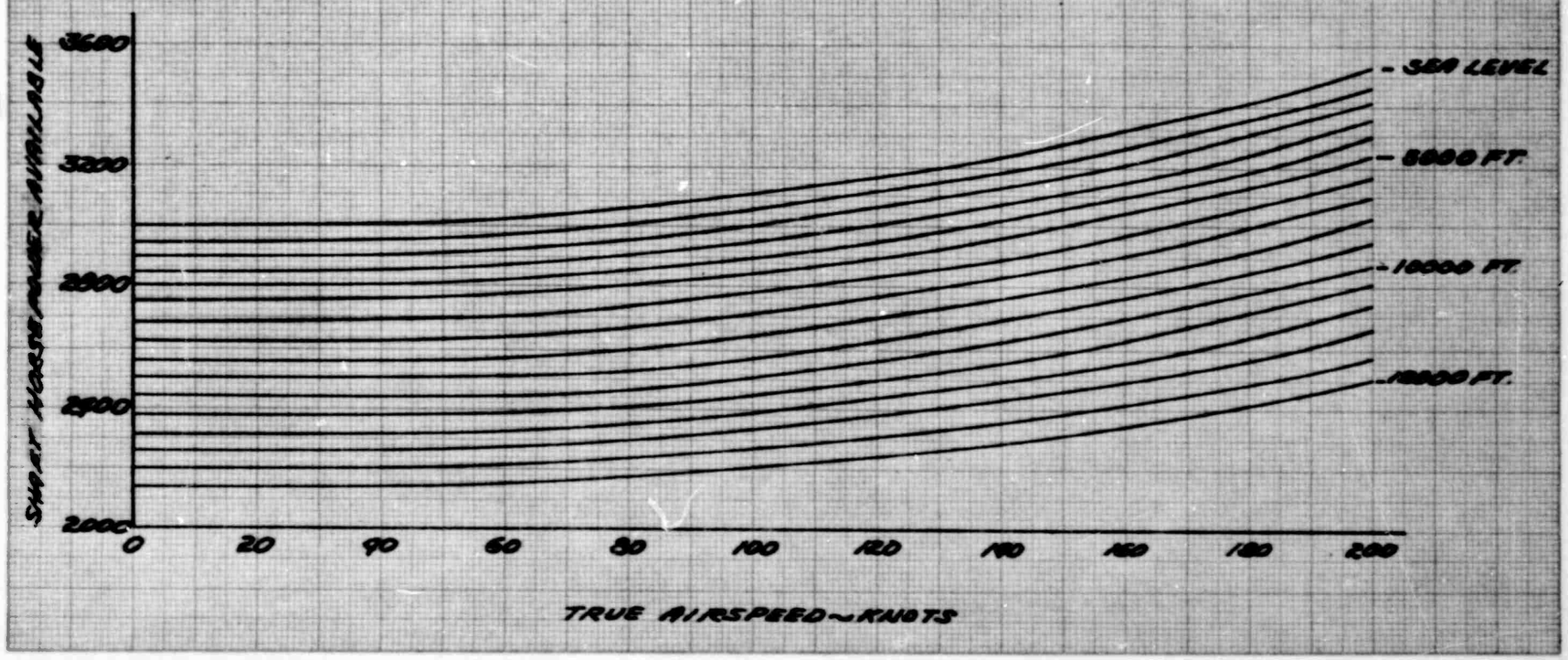


FIGURE 53

SHAFT HORSEPOWER AVAILABLE WITH RAM EFFECTS

CH-47C USA S/N 68-15859

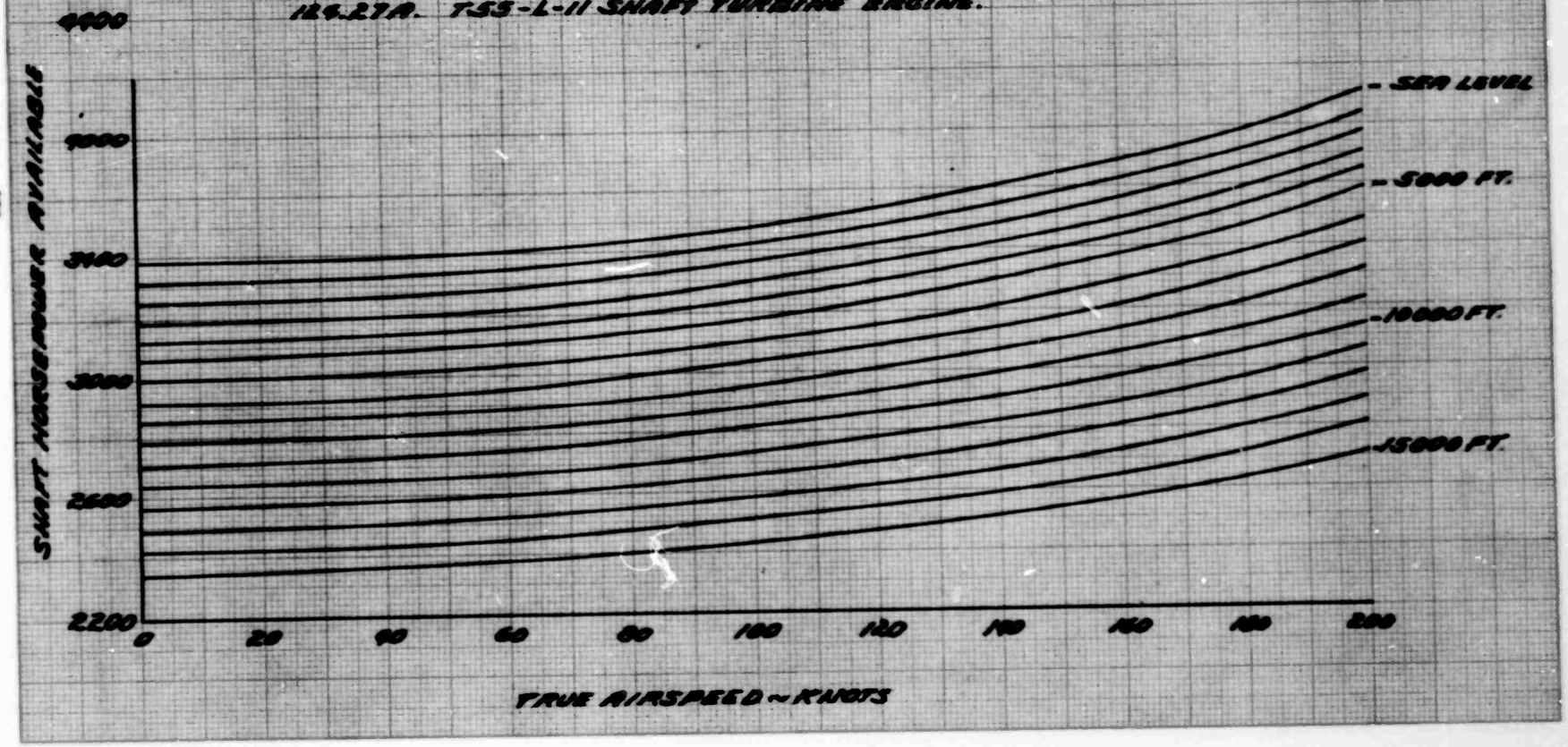
LYCOMING T53-L-11 ENGINE

MILITARY RATED POWER

ROTOR SPEED = 225 RPM

NOTES:

1. STANDARD DAY.
2. ENGINE INLET LOSSES BASED ON FIGURE 59 AND 60
3. BASED ON AVCO LYCOMING MODEL SPEC. NO. 126.27A. T53-L-11 SHAFT TURBINE ENGINE.



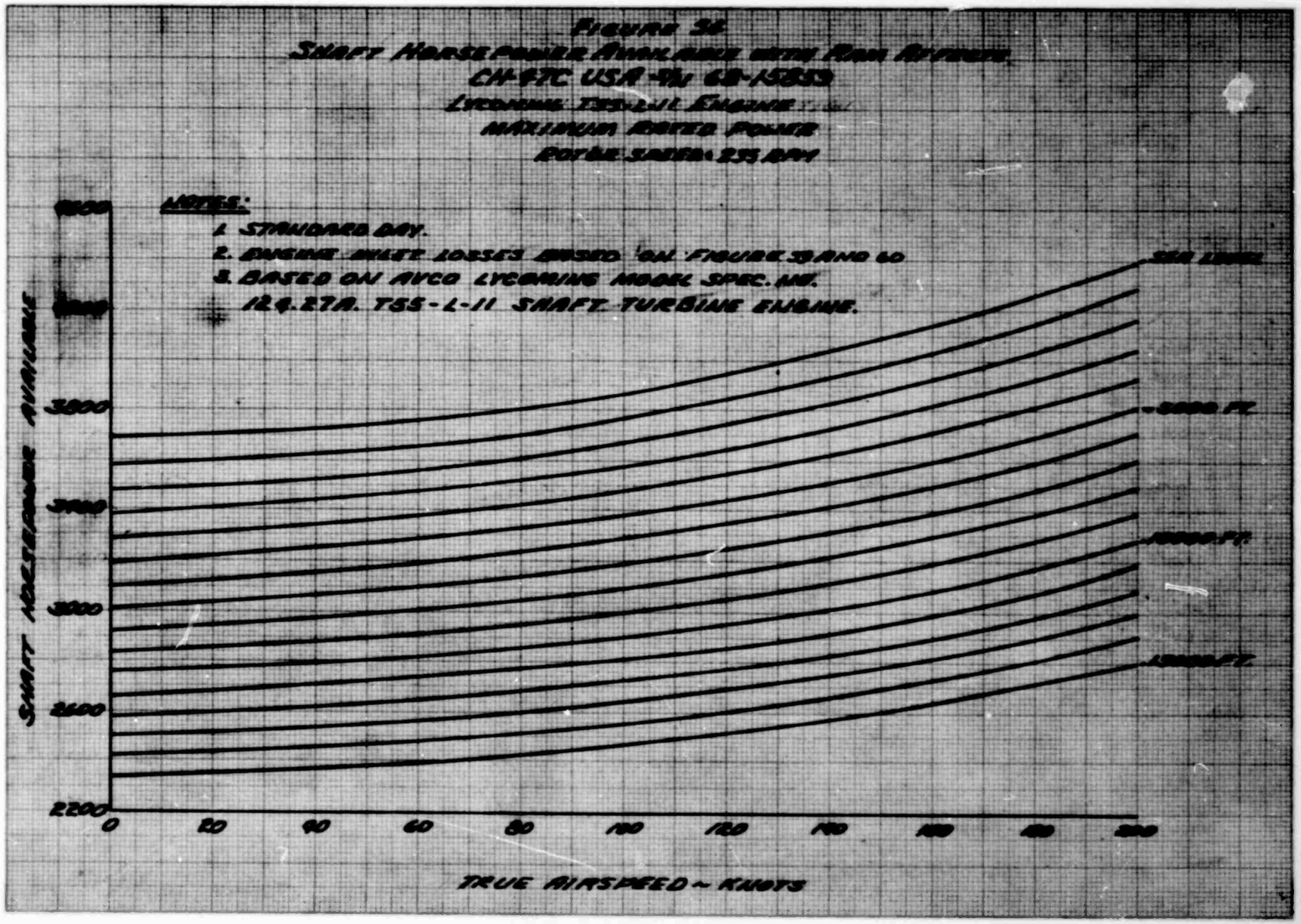


FIGURE 36
SHAFT HORSEPOWER AVAILABLE WITH RAM RECOVERY
CH-47C USA-YN 68-15859
LYCOMING T55-L-11 ENGINE
MAXIMUM RATED POWER
ROTOR SPEED 225 RPM
NOTES:
1. STANDARD DAY.
2. ENGINE INLET LOSSES BASED ON FIGURE 35 AND 60
3. BASED ON AVCO LYCOMING MODEL SPEC. NO. 124.27A. T55-L-11 SHAFT TURBINE ENGINE.
SHAFT HORSEPOWER AVAILABLE
3000
2600
2200
0
20
40
60
80
100
120
140
160
180
200
TRUE AIRSPEED ~ KNOTS

FIGURE 53
SHAFT HORSEPOWER AVAILABLE WITH RAM EFFECTS
CH-47C USA S/N 68-15859
LYCOMING T55-L-11 ENGINE
NORMAL RATED POWER
ROTOR SPEED = 225 RPM

NOTES:
1. STANDARD DAY.
2. ENGINE INLET LOSSES BASED ON FIGURE 59 AND 60
3. BASED ON AVCO LYCOMING MODEL SPEC. NO. 124.27A. T55-L-11 SHAFT TURBINE ENGINE.

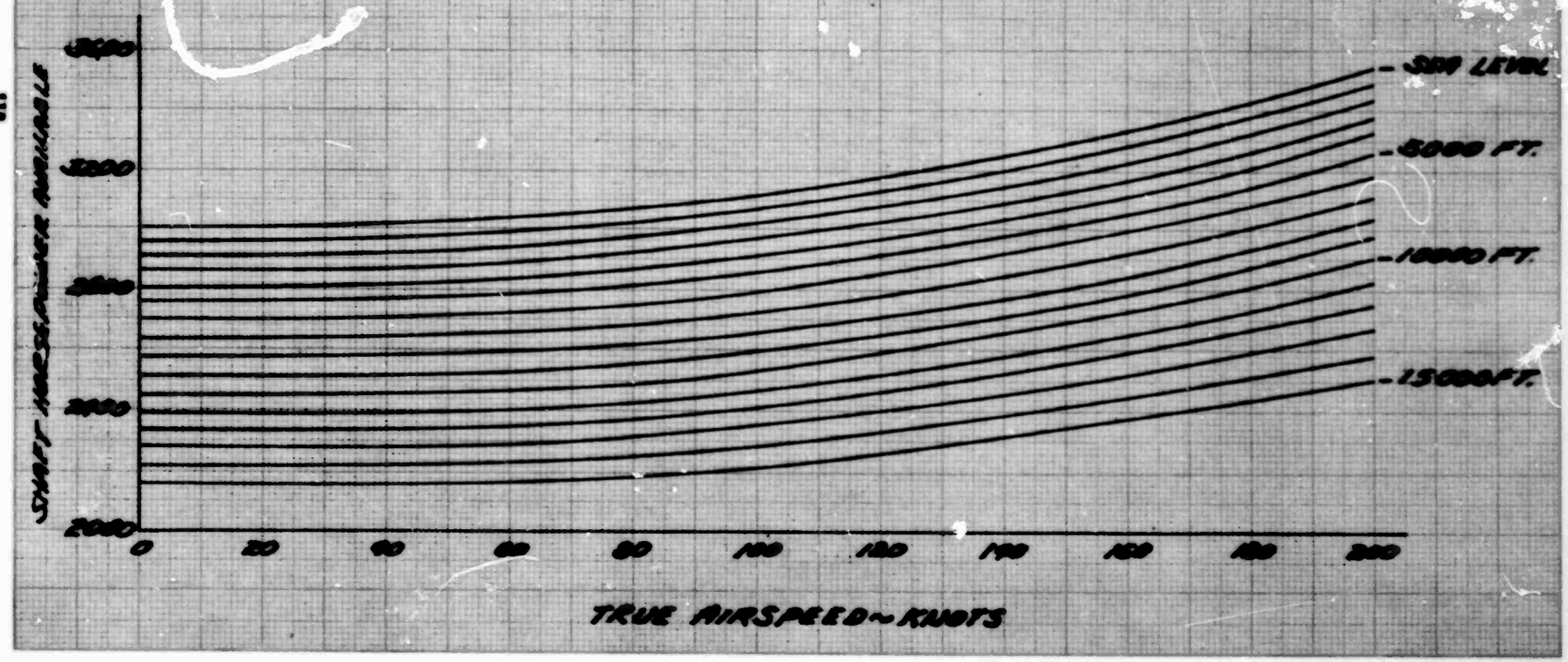


FIGURE 56

SHAFT HORSEPOWER AVAILABLE WITH RAM EFFECTS

CH-47C USA S/N 68-15859

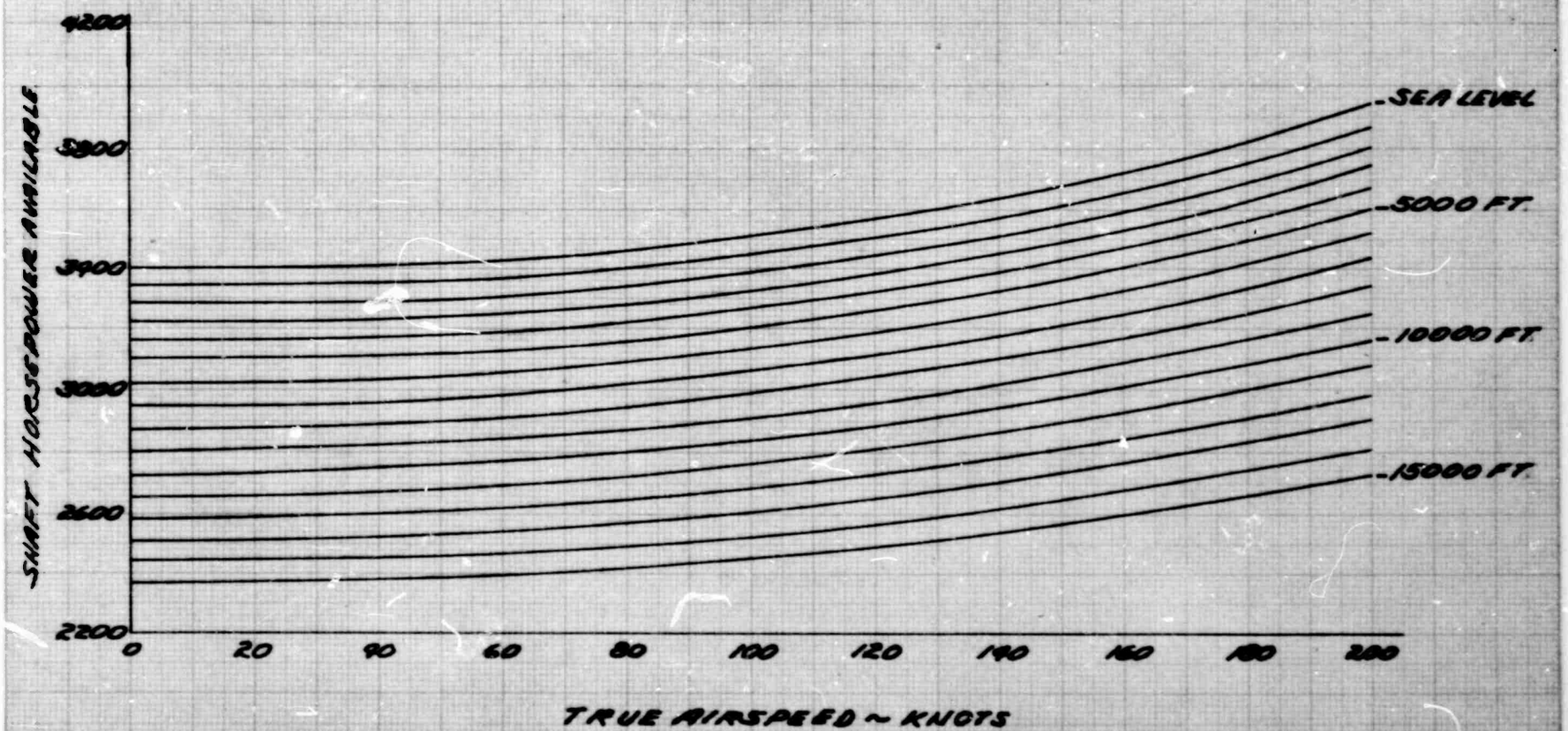
LYCOMING T55-L-11 ENGINE

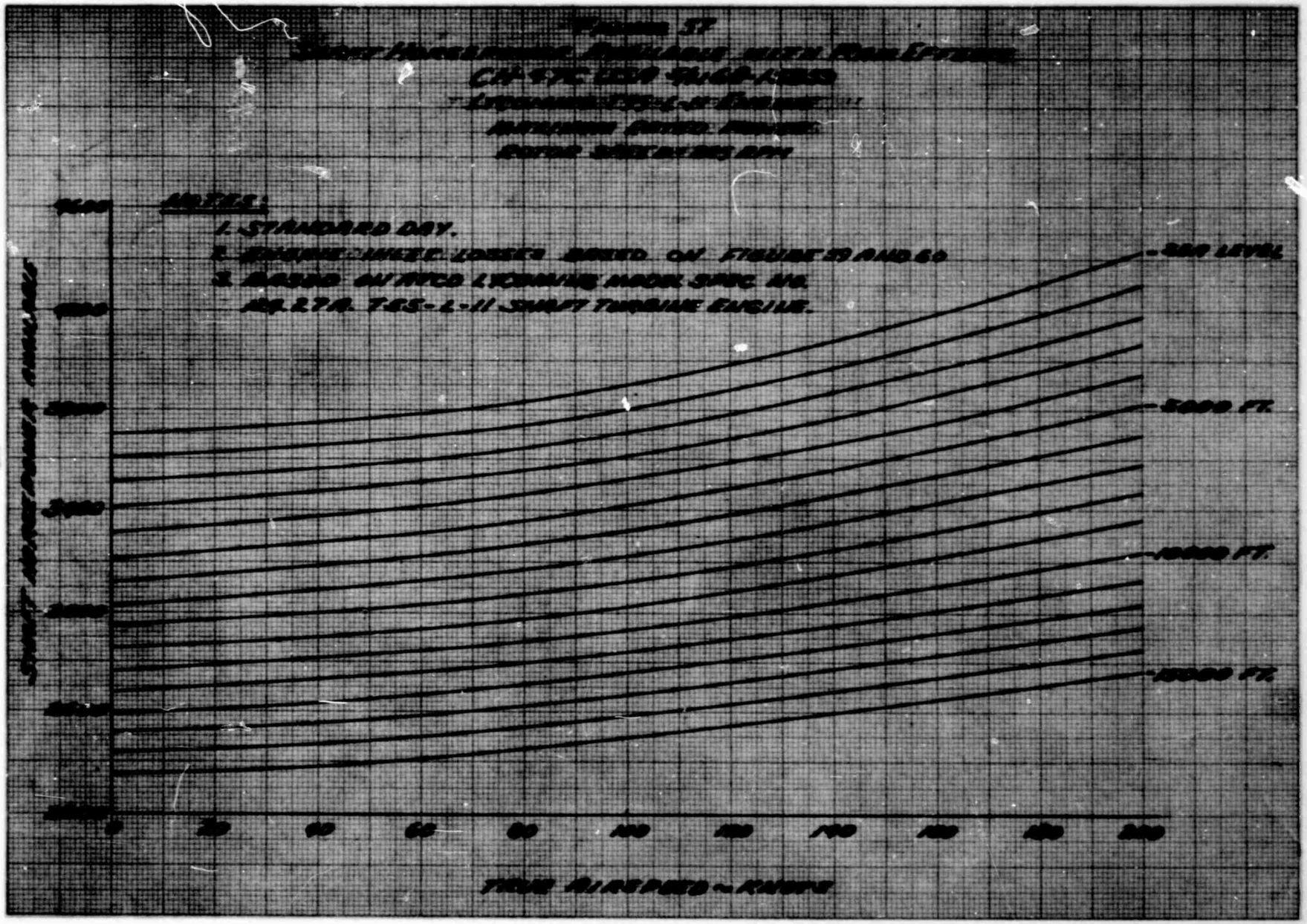
MILITARY RATED POWER

ROTOR SPEED = 245 RPM

NOTES:

1. STANDARD DAY.
2. ENGINE INLET LOSSES BASED ON FIGURE 59 AND 60
3. BASED ON AVCO LYCOMING MODEL SPEC. NO. T55-L-11 SHAFT TURBINE ENGINE.




NOTES:
1. STANDARD DAY.
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
TRUE AIRSPEED ~ KNOTS

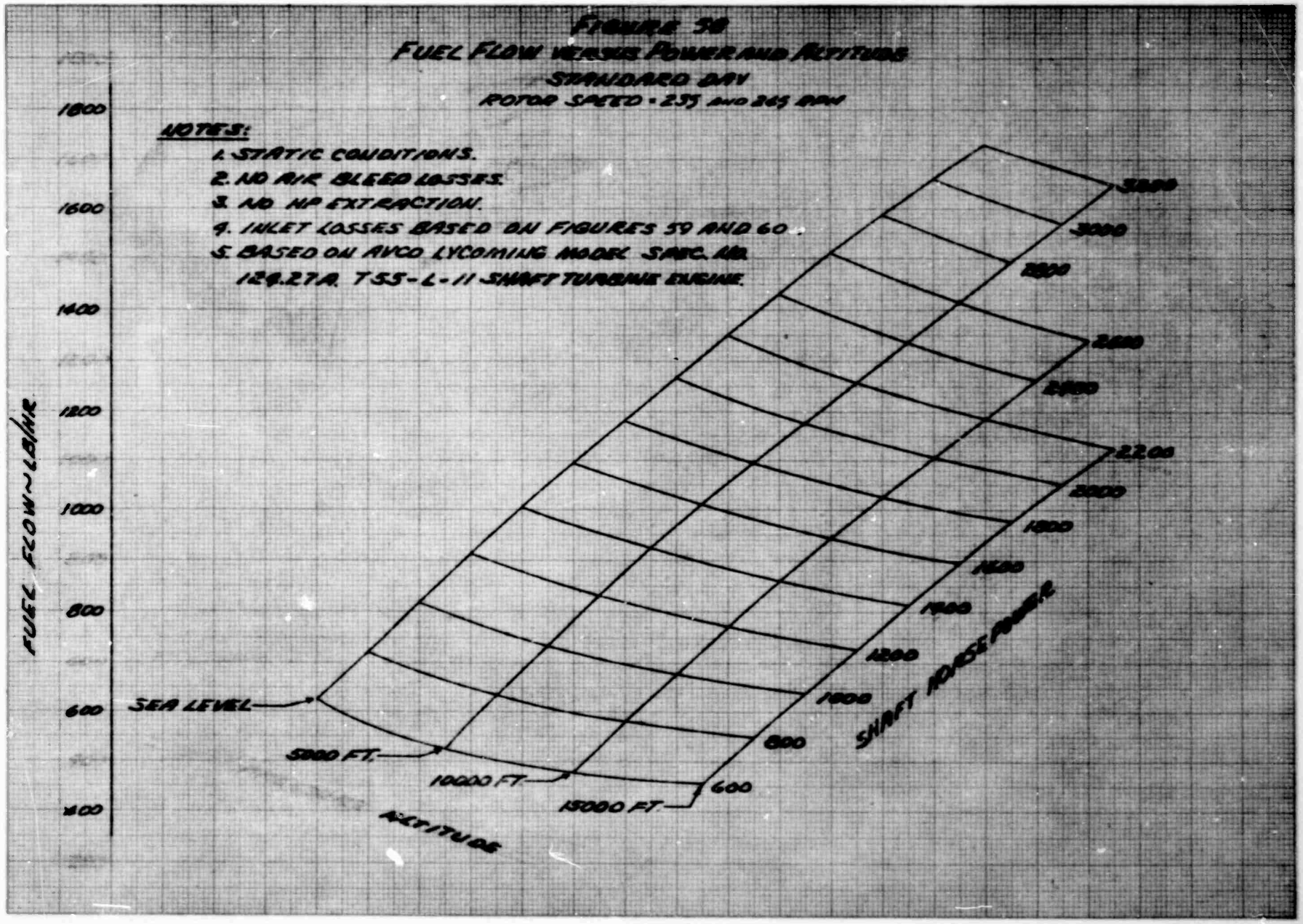


FIGURE 58
FUEL FLOW VERSUS POWER AND ALTITUDE
STANDARD DAY
ROTOR SPEED - 257 AND 265 RPM
NOTES:
1. STATIC CONDITIONS.
2. NO AIR BLEED LOSSES.
3. NO HP EXTRACTION.
4. INLET LOSSES BASED ON FIGURES 59 AND 60.
5. BASED ON AVCO LYCOMING MODEL SPEC. NO. 124.27A, T55-L-11 SHAFT TURBINE ENGINE.
FUEL FLOW ~ LB/HR
1800
1600
1400
1200
1000
800
600
400
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
ALTITUDE
SHAFT HORSE POWER
600
800
1000
1200
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200

FIGURE 59
ENGINE INLET CHARACTERISTICS
CH-47C USA S/N 68-15859
T-55-L-11 ENGINES

NOTE: 1. CURVE TAKEN FROM CH-47B (PHASE D) TECHNICAL REPORT NO. 66-23 MARCH 1970
2. ENGINE SCREENS OFF

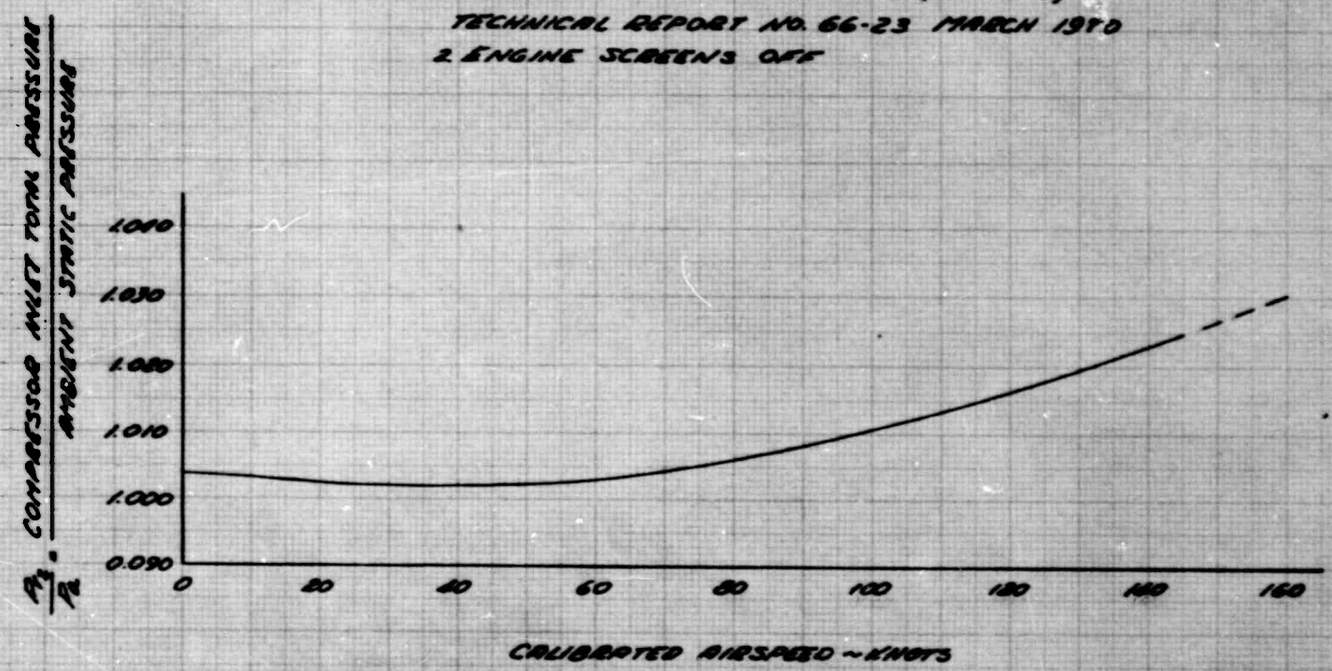
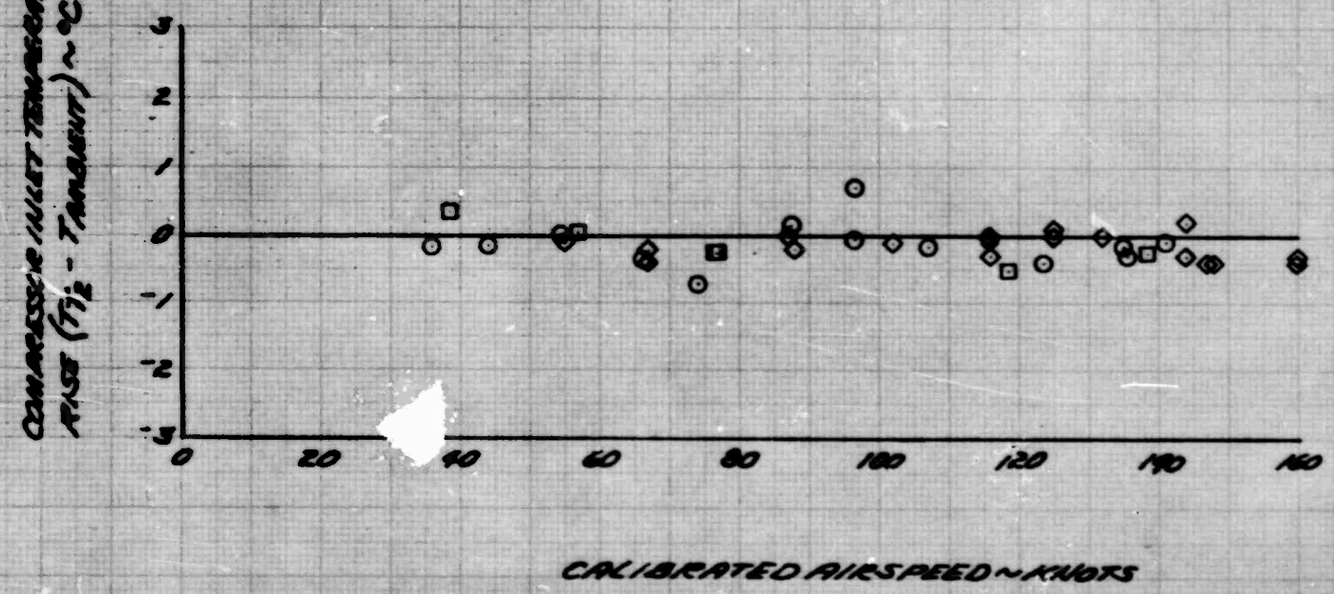


FIGURE 60
ENGINE INLET CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 ENGINES
ENGINE SCREENS OFF

SYMBOL	AVG. PRESSURE ALTITUDE ~ FT.	AVG. C.G. LOCATION ~ IN.	AVG. ROTOR SPEED ~ RPM
○	5750	331.0 (MID)	235
□	3690	330.7 (MID)	245
◇	1900	330.4 (MID)	235



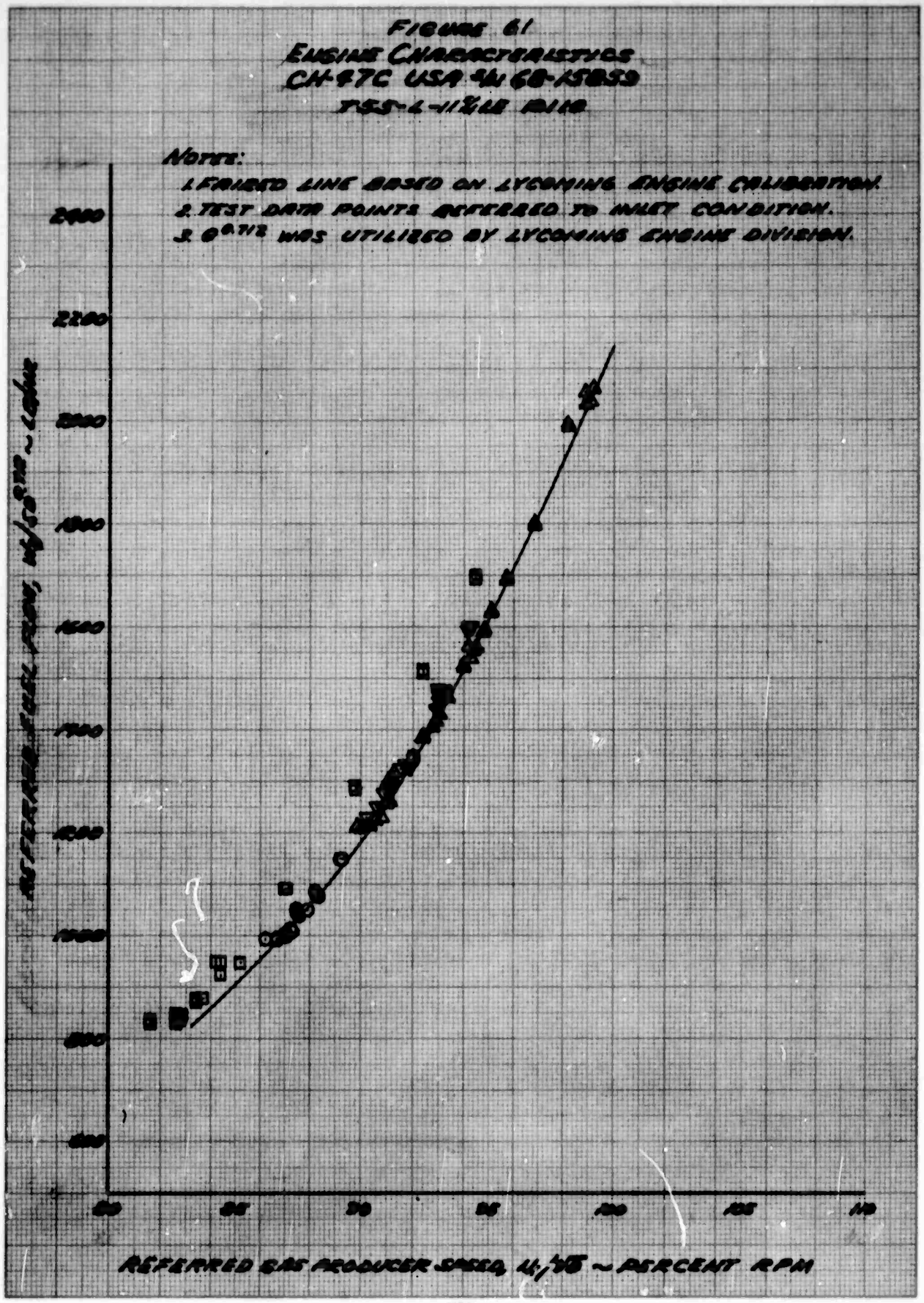


FIGURE 61
ENGINE CHARACTERISTICS
CH-47C USA 66-15859
T55-L-11 ENGINE
NOTES:
1. FAIRED LINE BASED ON LYCOMING ENGINE CALIBRATION.
2. TEST DATA POINTS REFERRED TO INLET CONDITION.
3. θ^0.712 WAS UTILIZED BY LYCOMING ENGINE DIVISION.
2400
2200
2000
1800
1600
1400
1200
1000
800
600
80
85
90
95
100
105
110
REFERRED GAS PRODUCER SPEED, N/√θ ~ PERCENT RPM

FIGURE 62
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
TSE-L-H-%LE 19110

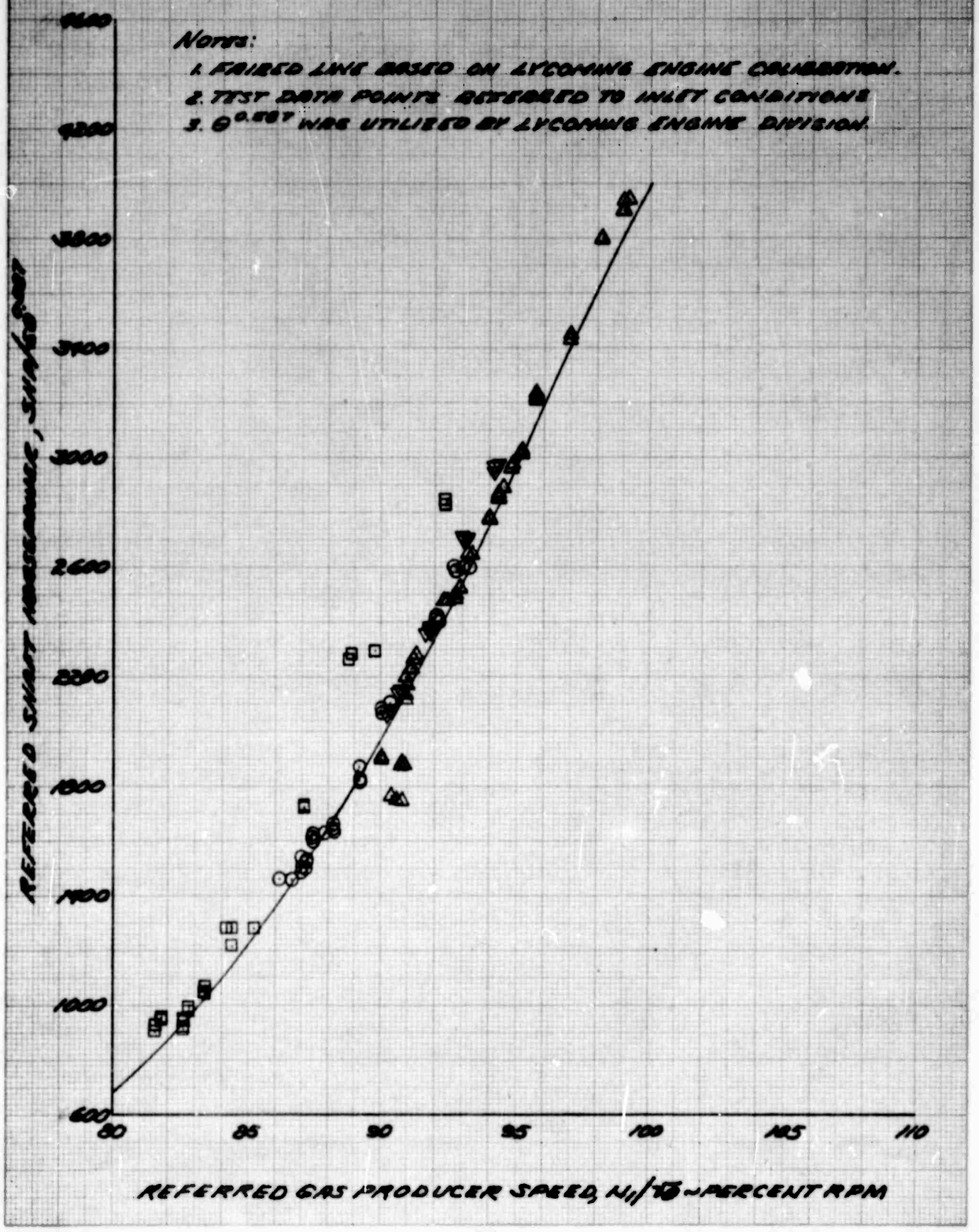


FIGURE 63
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 & LE 19110

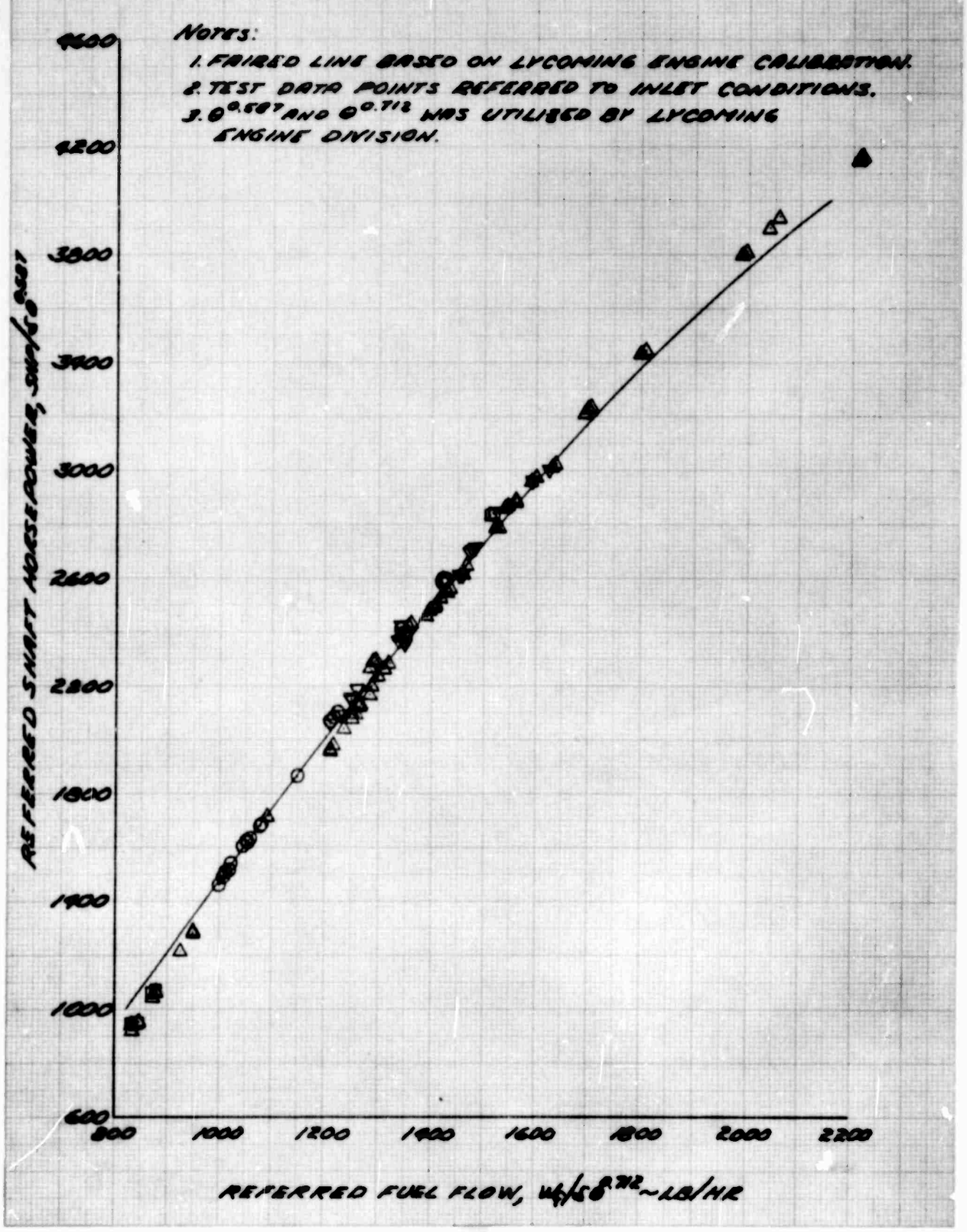


FIGURE 64
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
TSS-L-11 & LE 19166

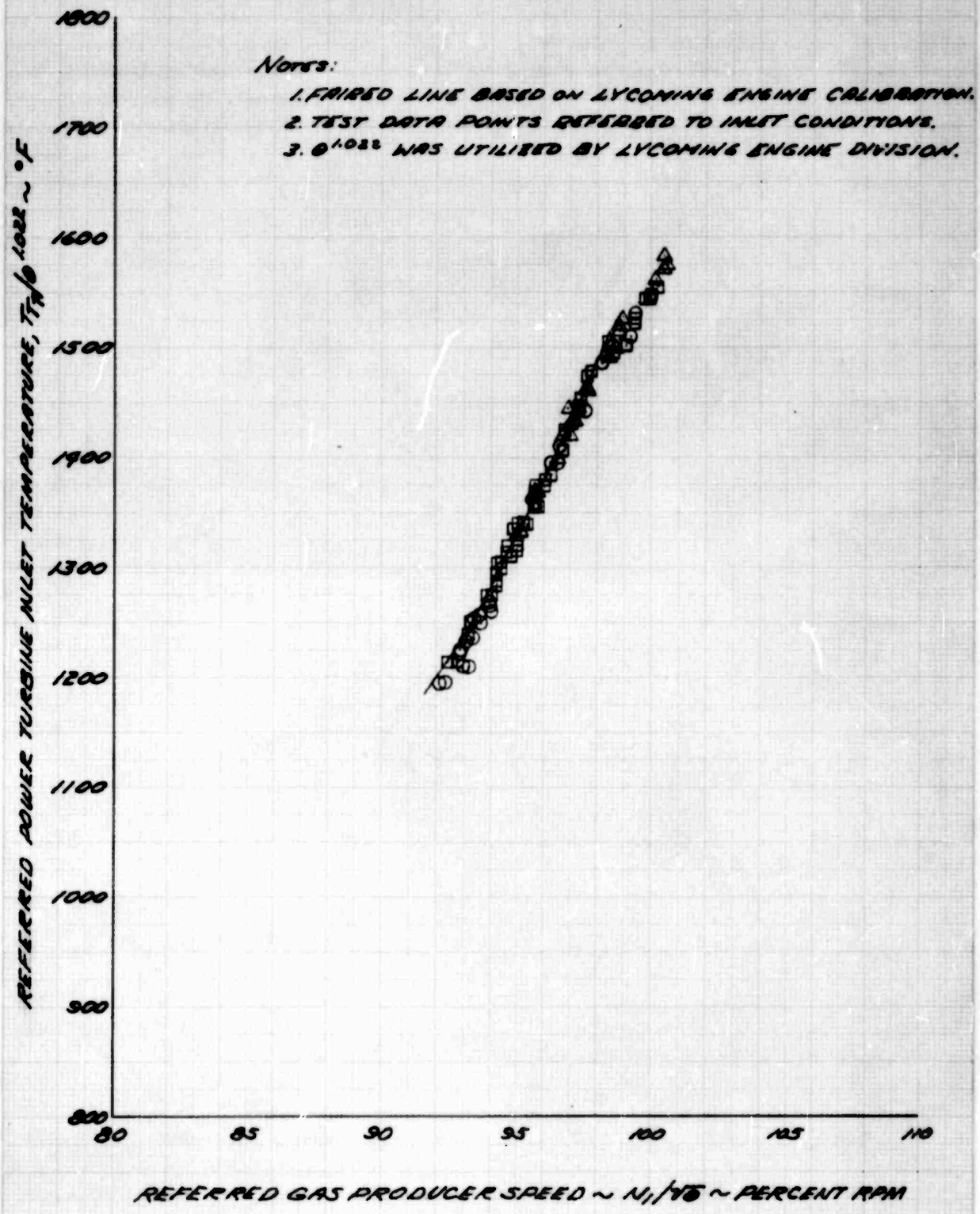


FIGURE 65
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 ½LE 19196

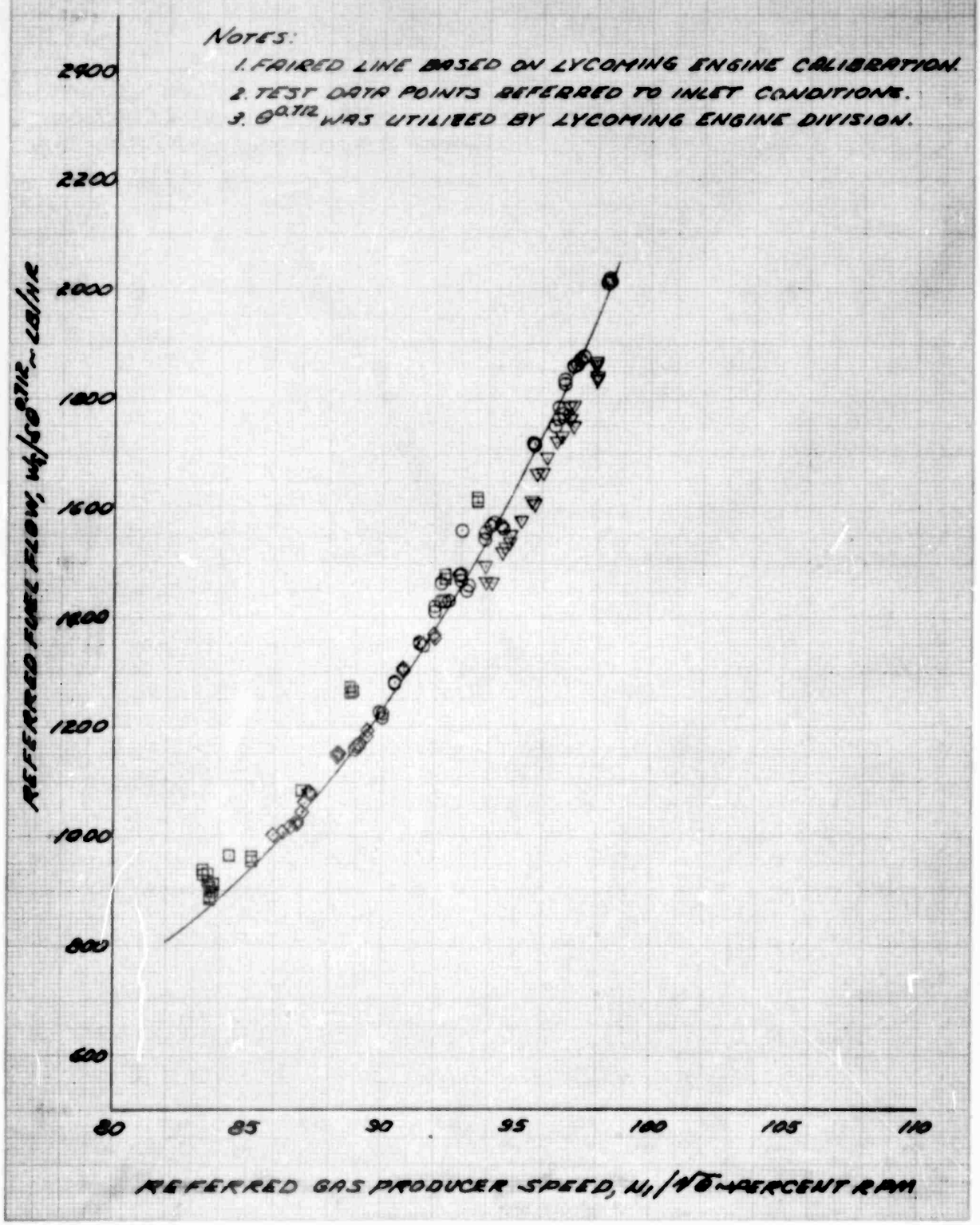


FIGURE 66
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
TSS-L-11 & LE 19196

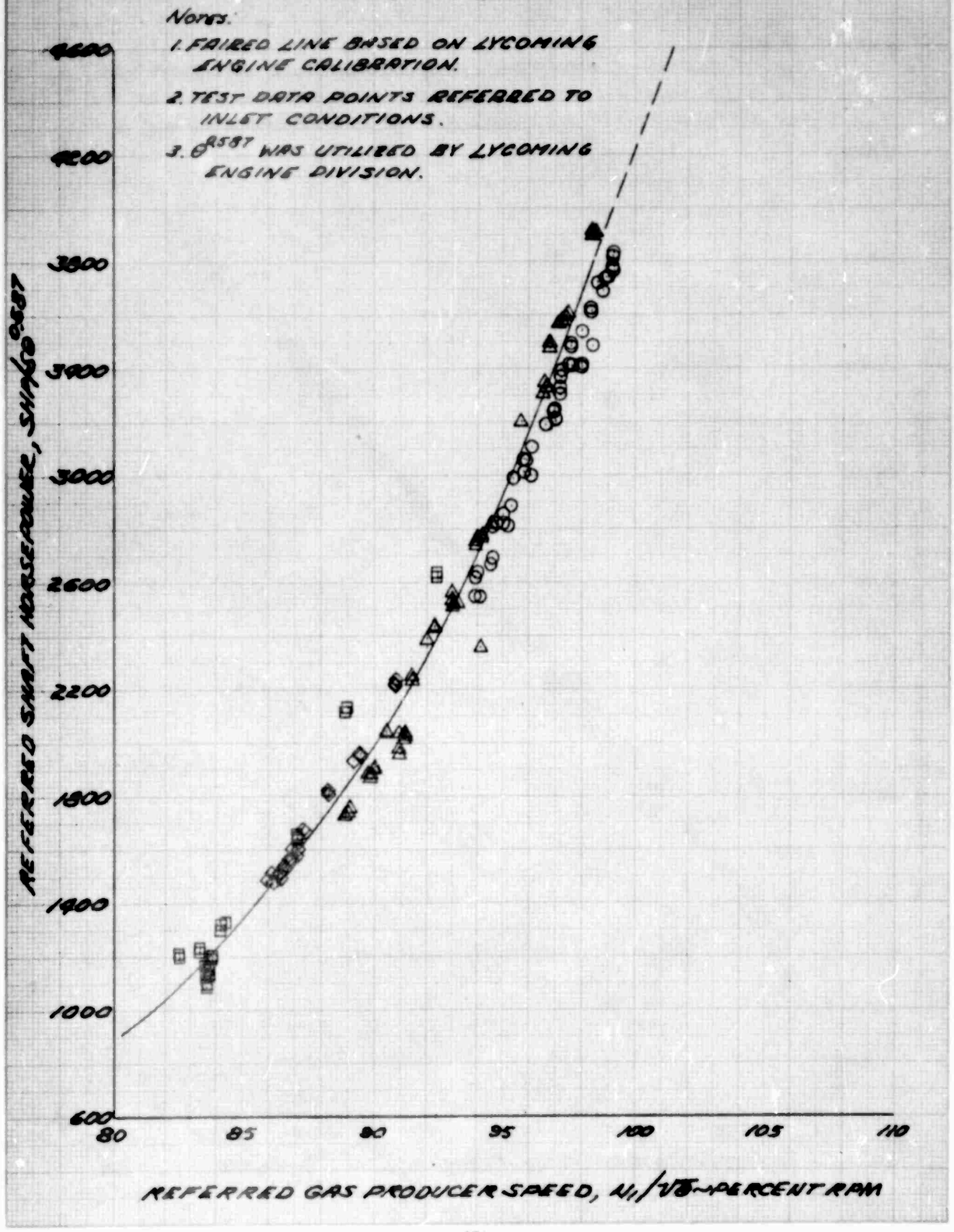


FIGURE 67
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 & LE 19196

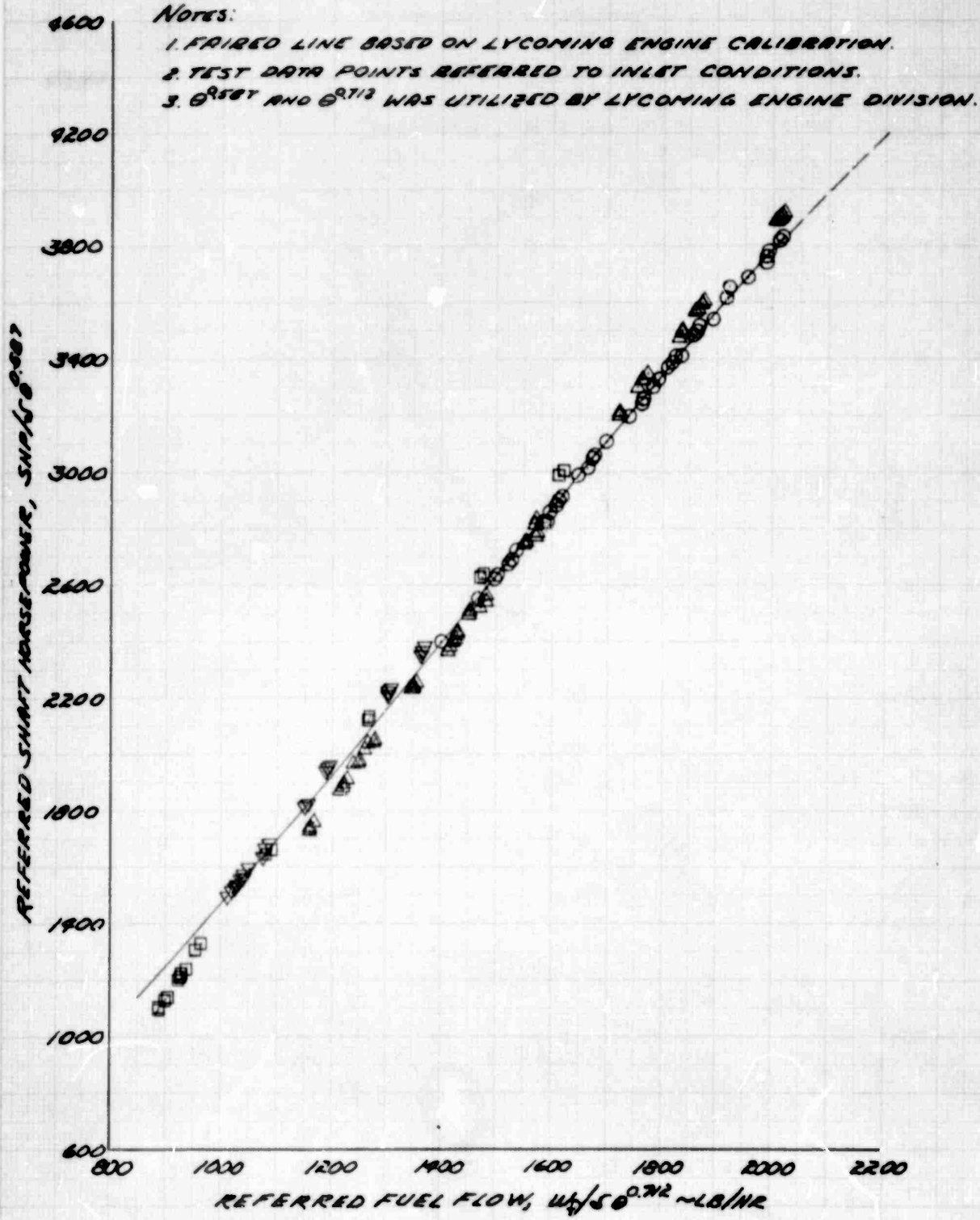


FIGURE 68
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 S/N LE 19258

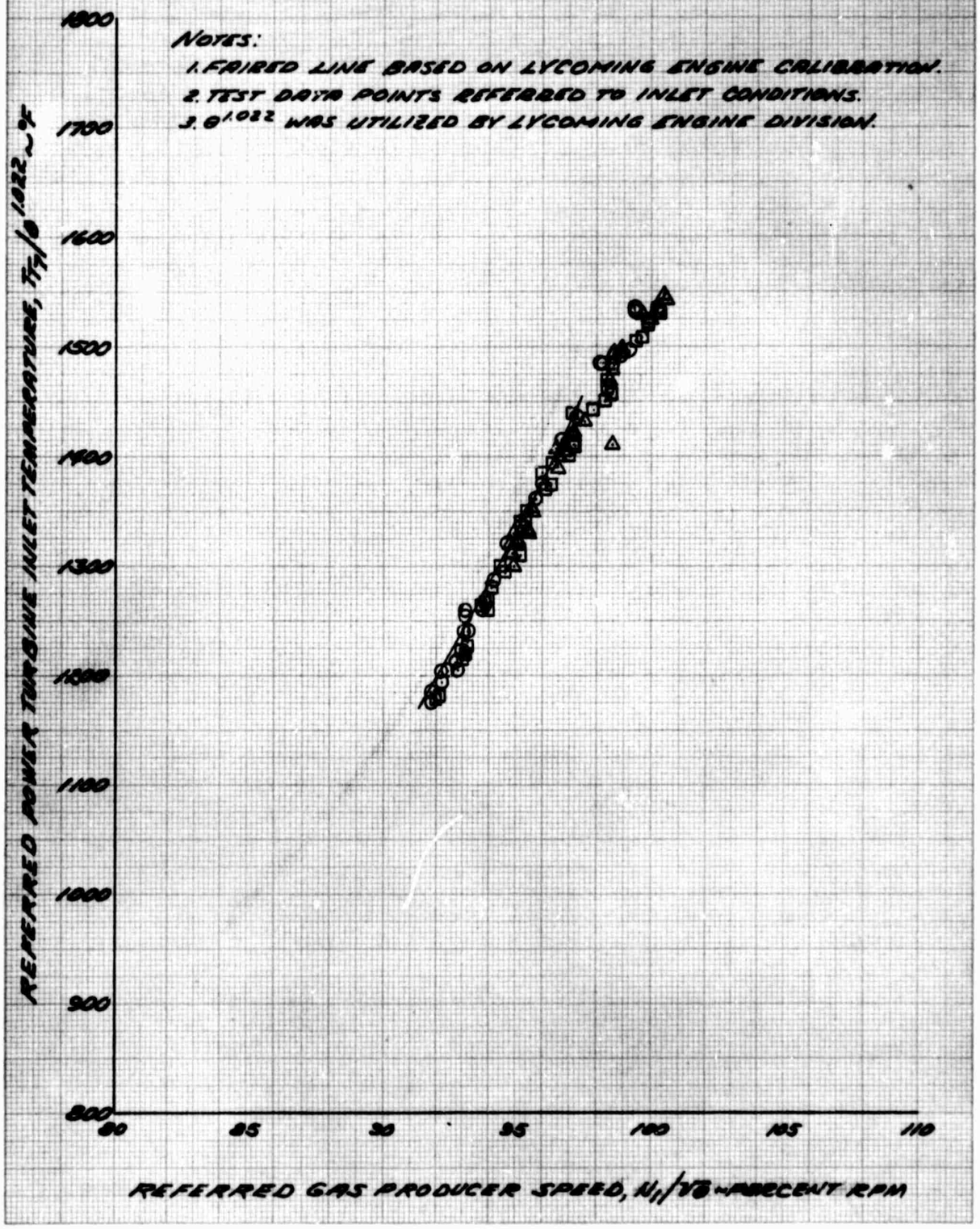


FIGURE 69
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15853
T55-L-11 & LE 19258

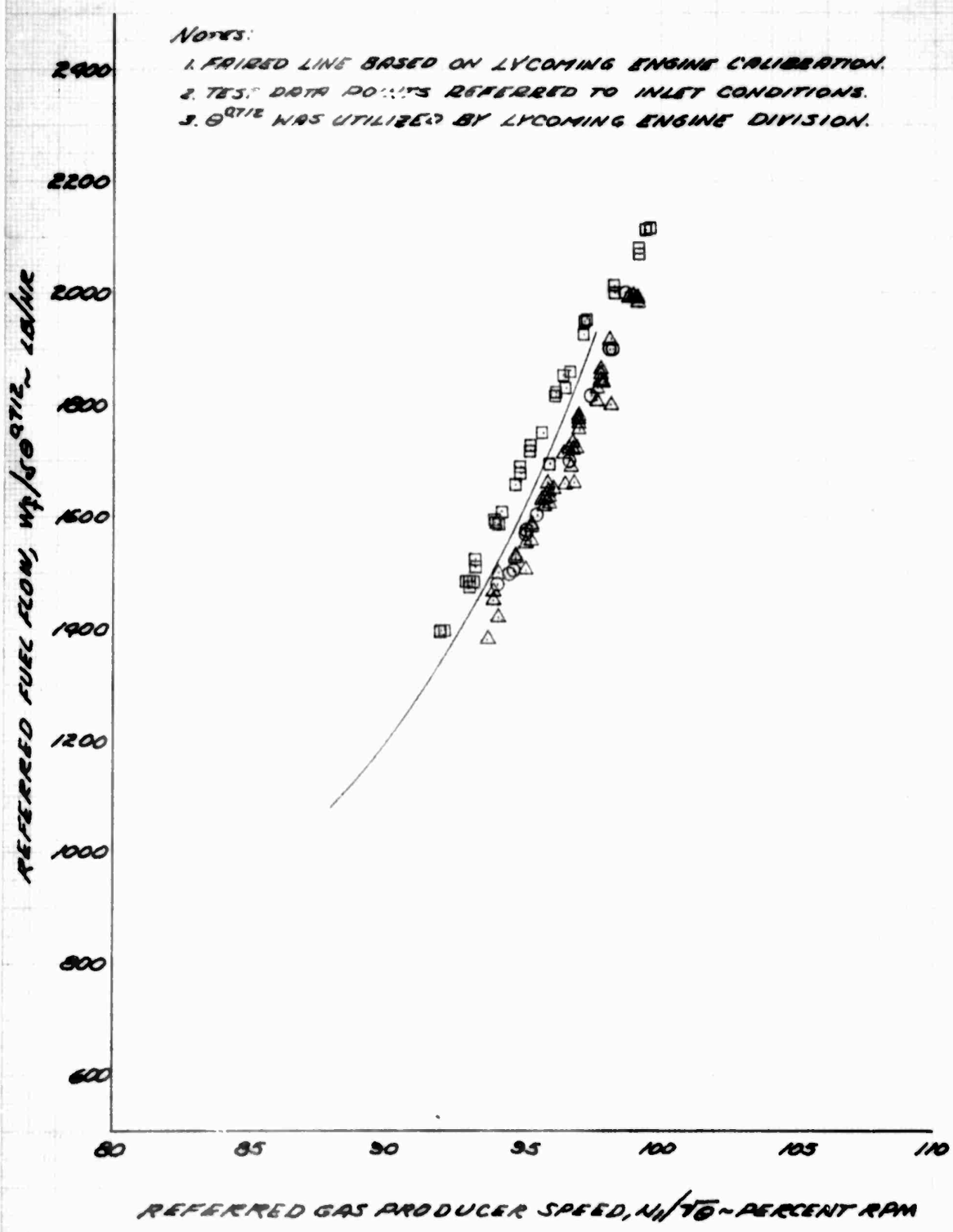


FIGURE 70
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 S/LE 19858

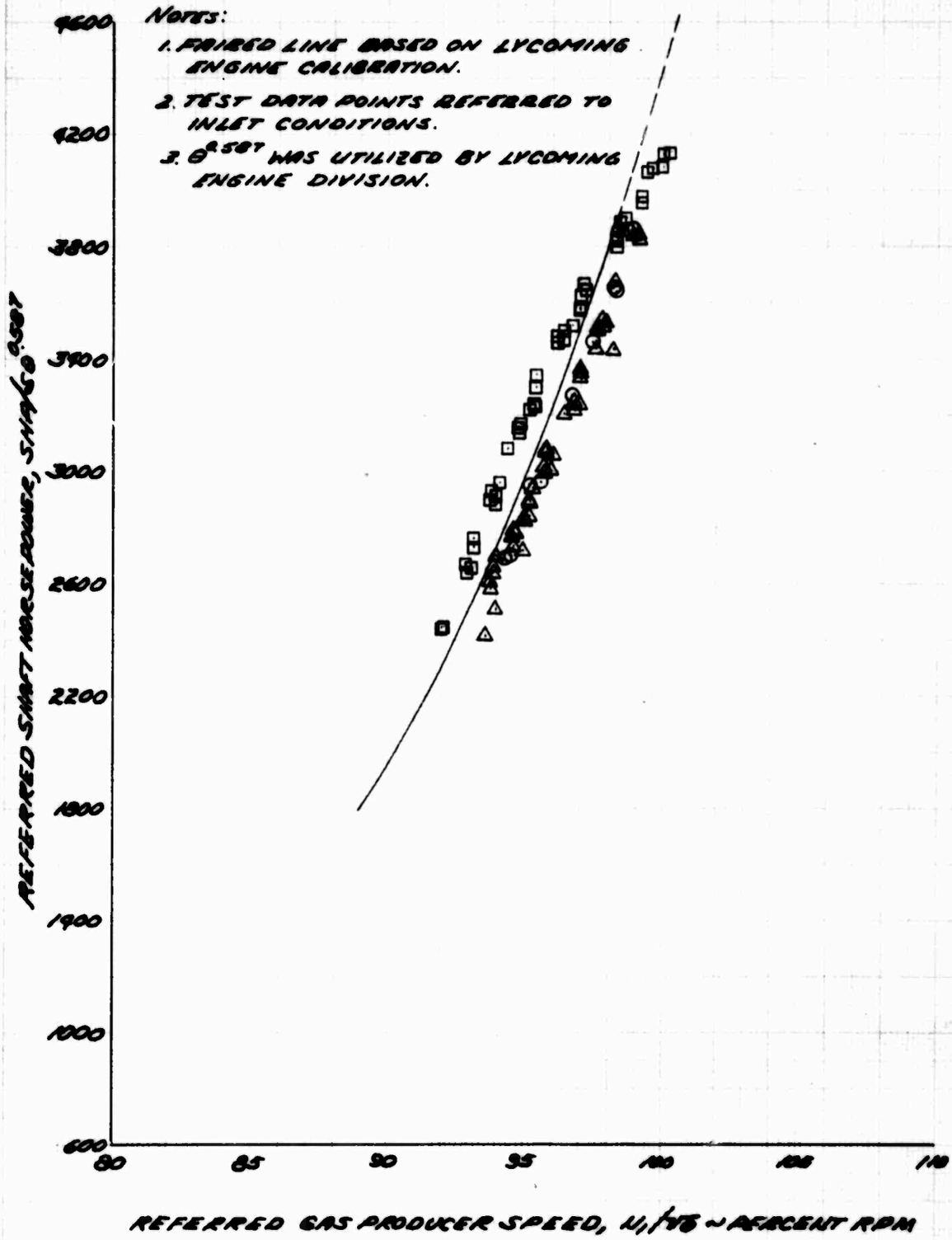
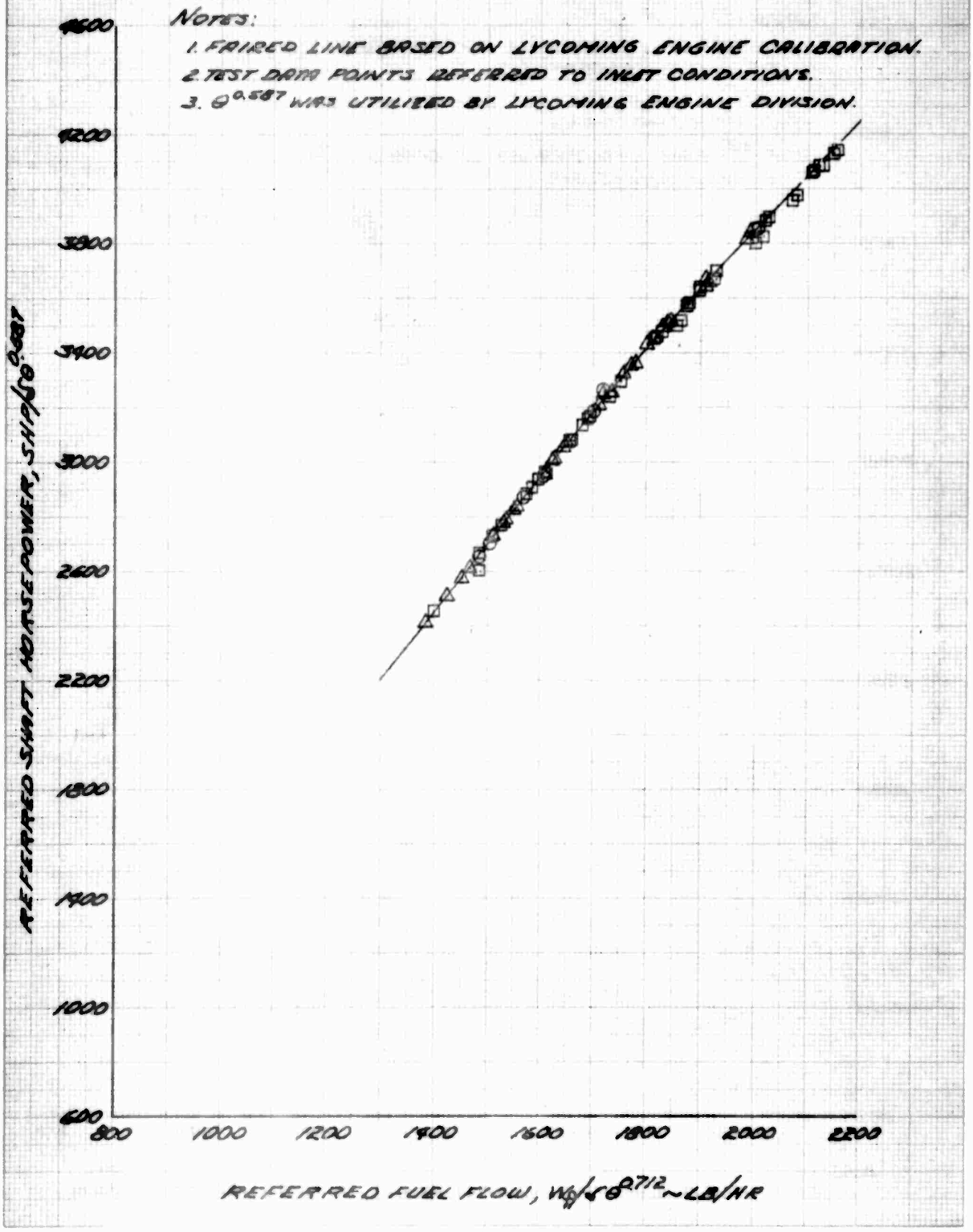


FIGURE 71
ENGINE CHARACTERISTICS
CH-47C USA S/N 68-15859
T55-L-11 S/N LE 19258



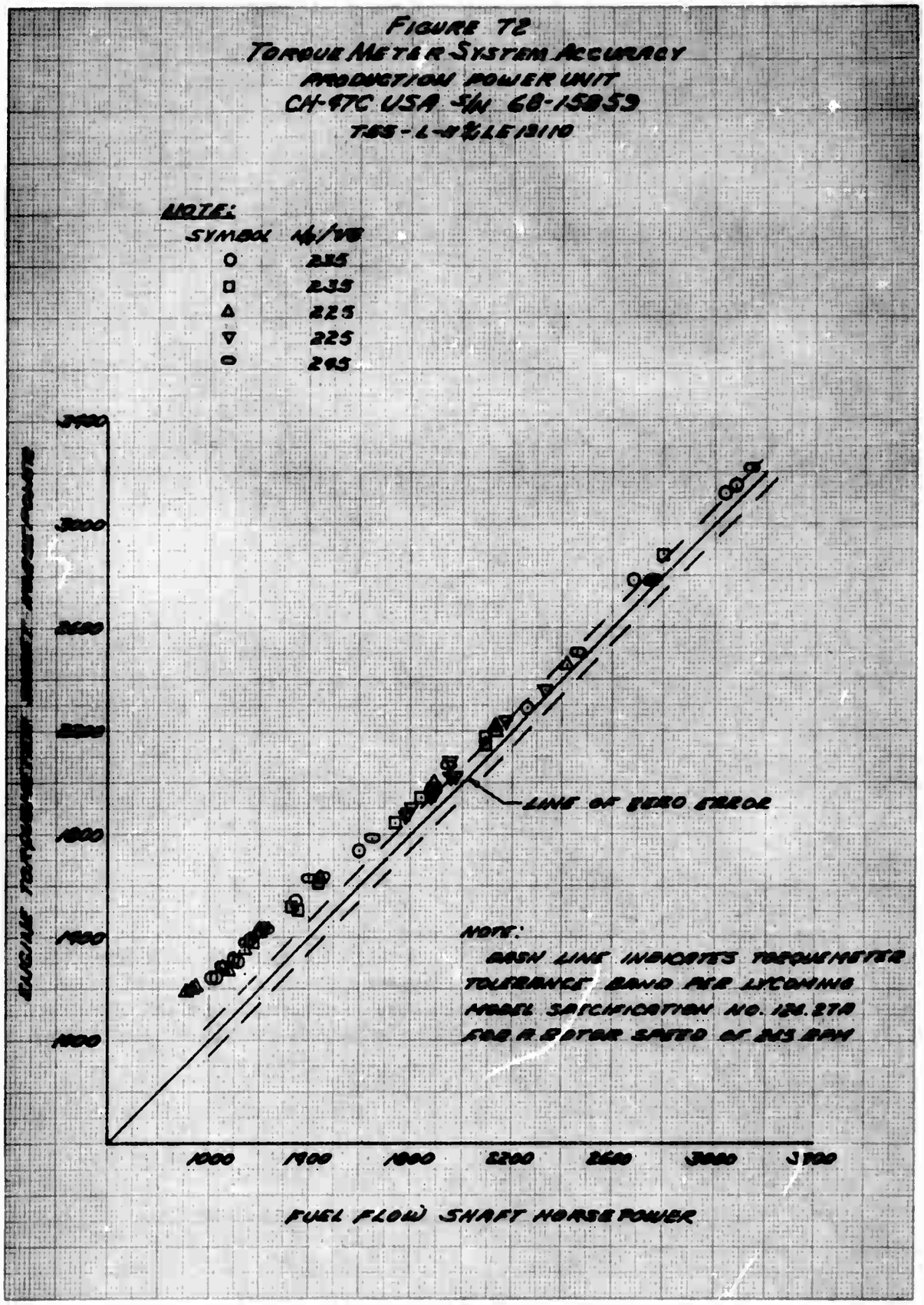
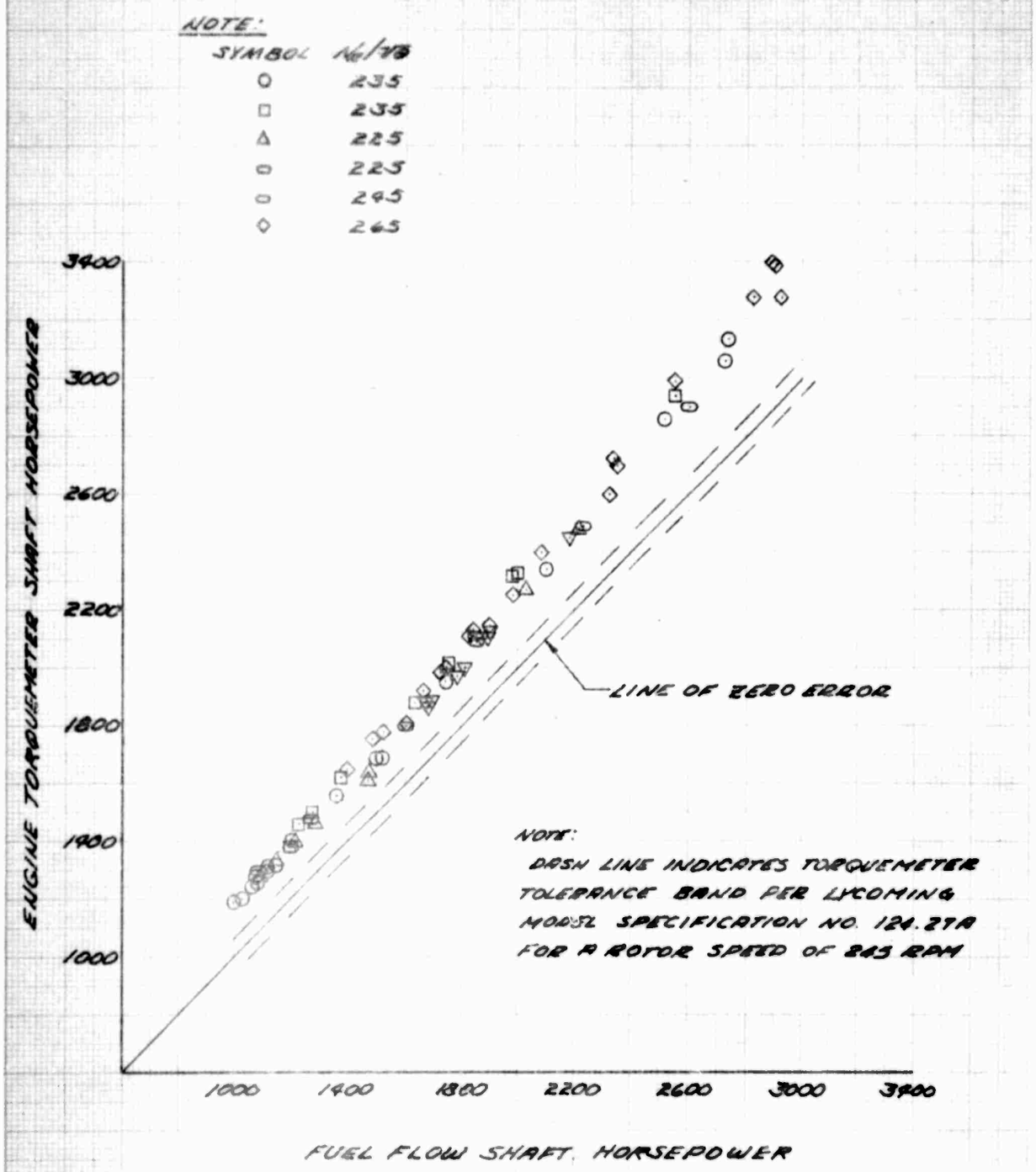


FIGURE 72
TORQUE METER SYSTEM ACCURACY
PRODUCTION POWER UNIT
CH-47C USA S/N 68-15859
NOTE:
SYMBOL
235
235
225
225
245
LINE OF ZERO ERROR
NOTE:
DASH LINE INDICATES TORQUEMETER
TOLERANCE BAND PER LYCOMING
MODEL SPECIFICATION NO. 126.27A
FOR A ROTOR SPEED OF 245 RPM
1000
1400
1800
2200
2600
3000
3400
FUEL FLOW SHAFT HORSEPOWER

FIGURE 73
TORQUE METER SYSTEM ACCURACY
PRODUCTION POWER UNIT
CH-47C USA S/N 68-15859
T55-L-11 & LE 19196



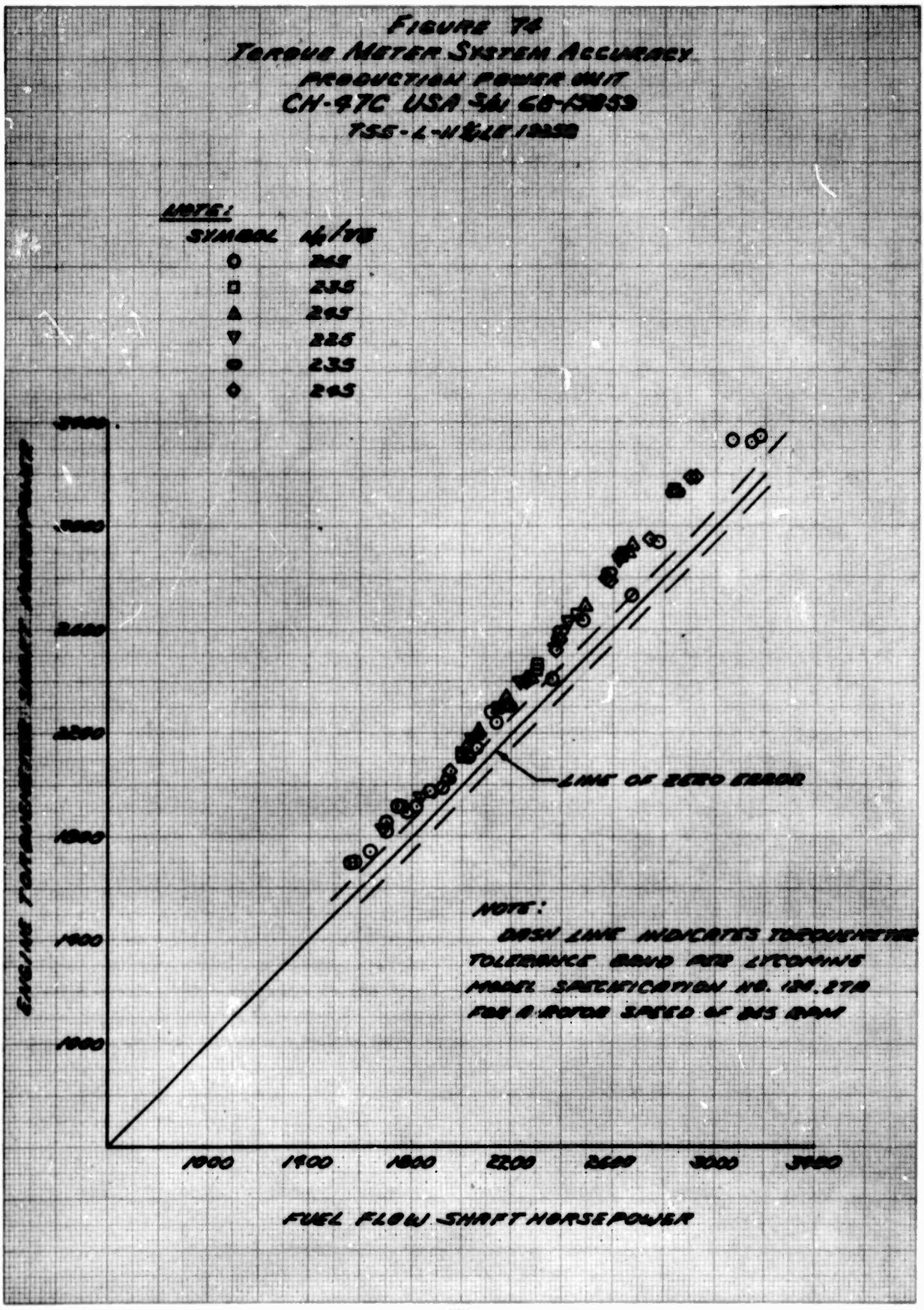


FIGURE 74
TORQUE METER SYSTEM ACCURACY
PRODUCTION POWER UNIT
CH-47C USA S/N 68-15859
NOTE:
SYMBOL
265
235
245
225
235
245
LINE OF ZERO ERROR
NOTE:
DASH LINE INDICATES TORQUEMETER
TOLERANCE BAND PER LYCOMING
MODEL SPECIFICATION NO. 124.27A
FOR A ROTOR SPEED OF 265 RPM
ENGINE TORQUEMETER SHAFT HORSEPOWER
1000
1400
1800
2200
2600
3000
3400
FUEL FLOW SHAFT HORSEPOWER

FIGURE 75
TORQUE METER SYSTEM ACCURACY
OPTIMIZED POWER UNIT
CH-47C USA S/N 68-15859
T55-L-11 W/LE 19110

NOTE:
$N_2/\sqrt{\theta} = 235$

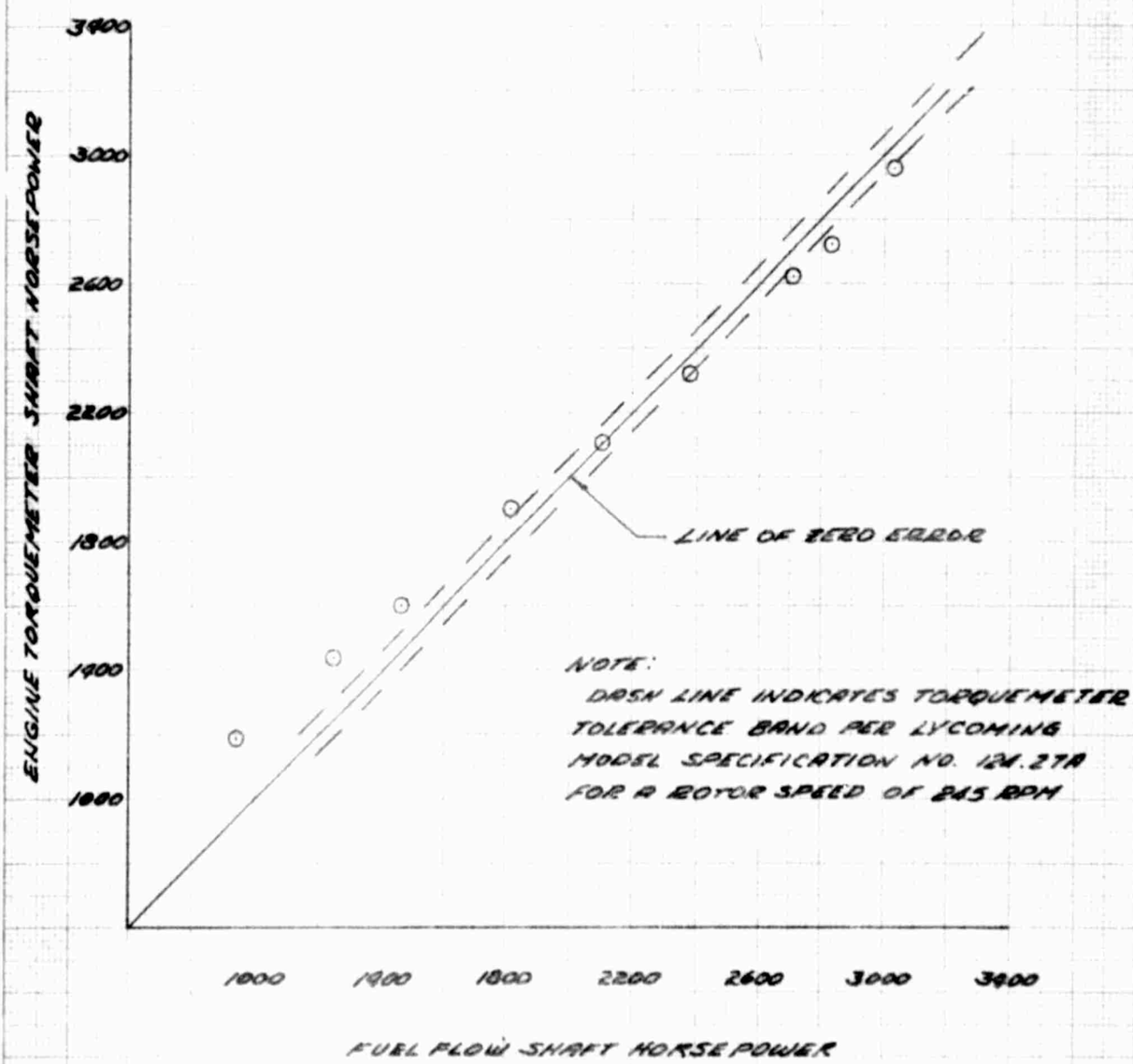
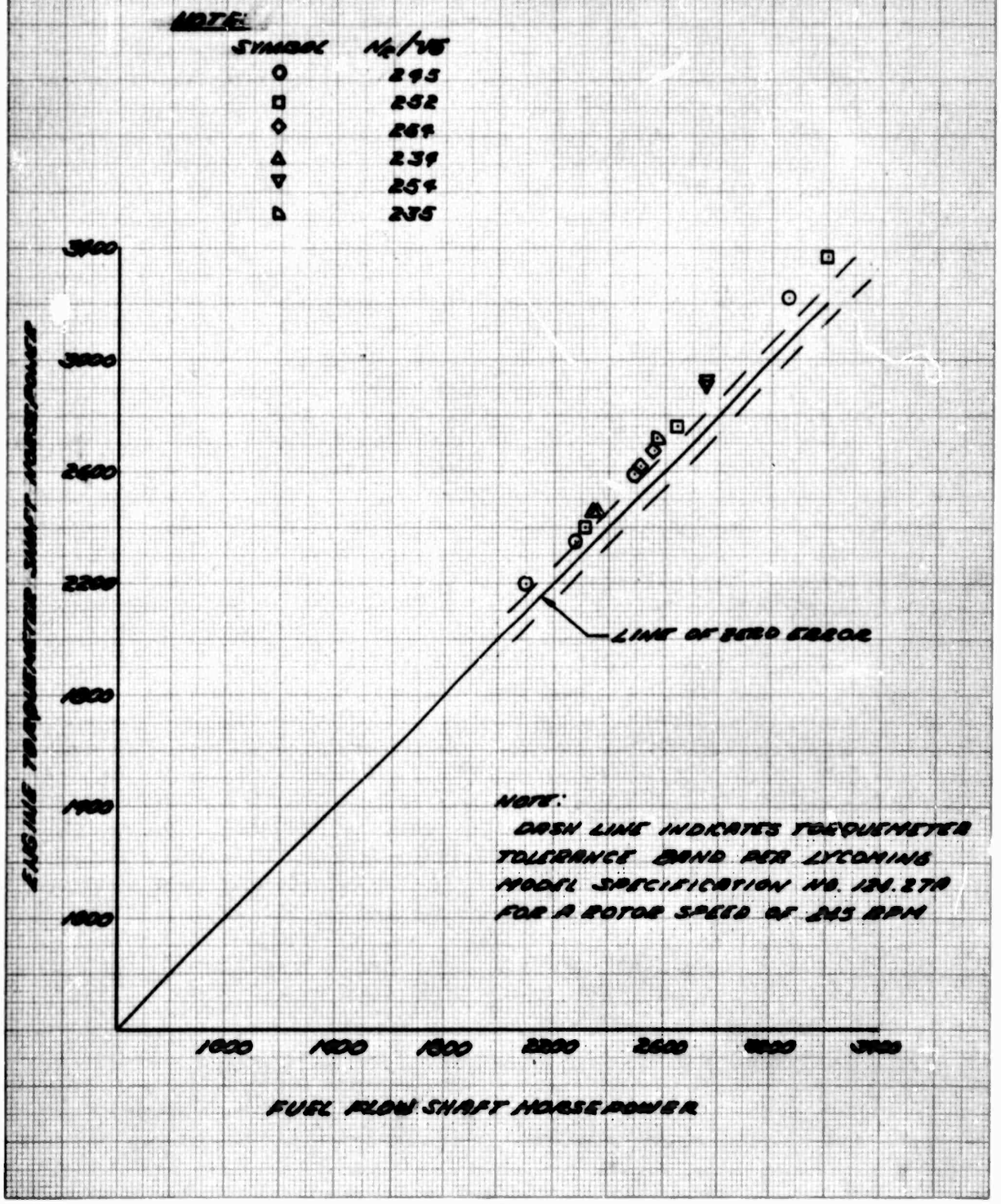


FIGURE 76
TORQUE METER SYSTEM ACCURACY
OPTIMIZED POWER UNIT
CH-47C USA S/N 68-15859
TSS-L-11 %LE 19/96



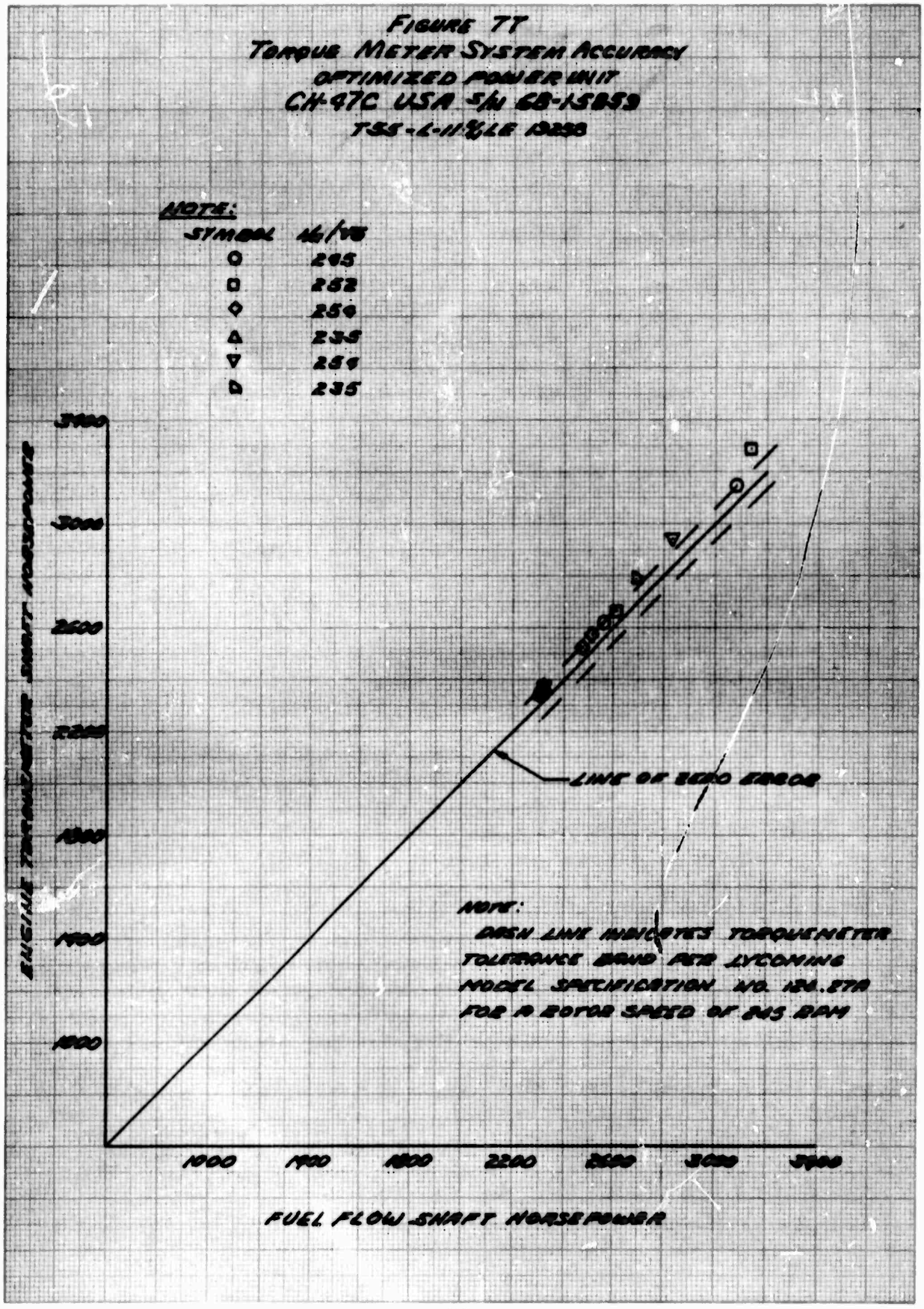


FIGURE 77
TORQUE METER SYSTEM ACCURACY
OPTIMIZED POWER UNIT
CH-47C USA S/N 68-15859
T55-L-11 S/N LE 19258
NOTE:
SYMBOL
245
252
254
235
254
235
LINE OF ZERO ERROR
NOTE:
DASH LINE INDICATES TORQUEMETER
TOLERANCE BAND PER LYCOMING
MODEL SPECIFICATION NO. 126.27A
FOR A ROTOR SPEED OF 245 RPM
ENGINE TORQUEMETER SHAFT HORSEPOWER
3400
3000
2600
2200
1800
1400
1000
1000
1400
1800
2200
2600
3000
3400
FUEL FLOW SHAFT HORSEPOWER

FIGURE 78
CRUISE GUIDE INDICATOR RESPONSE
WITH AIRSPEED
CH-47C USA S/N 68-15859
LEVEL FLIGHT

GROSS WEIGHT	CENTER OF GRAVITY	PRESSURE ALTITUDE	OAT	ROTOR SPEED
35,500 LB	MID	5640 FT	6.1°C	230 RPM

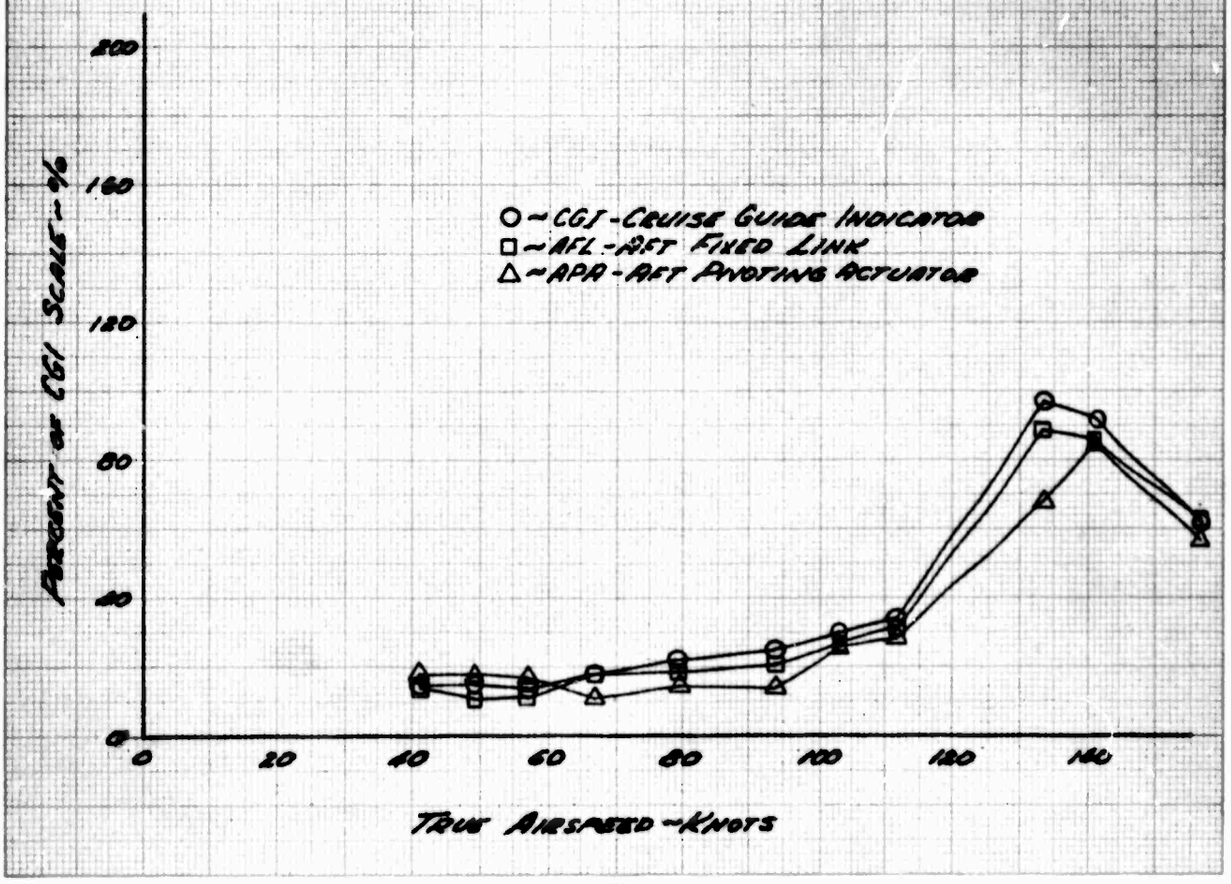


FIGURE 79
AIRSPEED CALIBRATION
CH-47C USA S/N 68-15859
BOOM SYSTEM IN LEVEL FLIGHT

SYMBOL	FLIGHT CONDITION	AVG. DENSITY ALTITUDE~FT	AVG. GROSS WEIGHT~LB	AVG. C.G. LOCATION~IN	AVG. ROTOR SPEED~RPM
○	LEVEL	5750	29730	331.0(MID)	235 (TRAIL. [illegible])
□	LEVEL	5690	28630	330.7(MID)	245 (TRAIL. [illegible])
◇	LEVEL	1900	27920	330.4(MID)	235 (GROUND [illegible])
△	LEVEL	4000	37500	330.0(MID)	235 AND 245 (PACE)

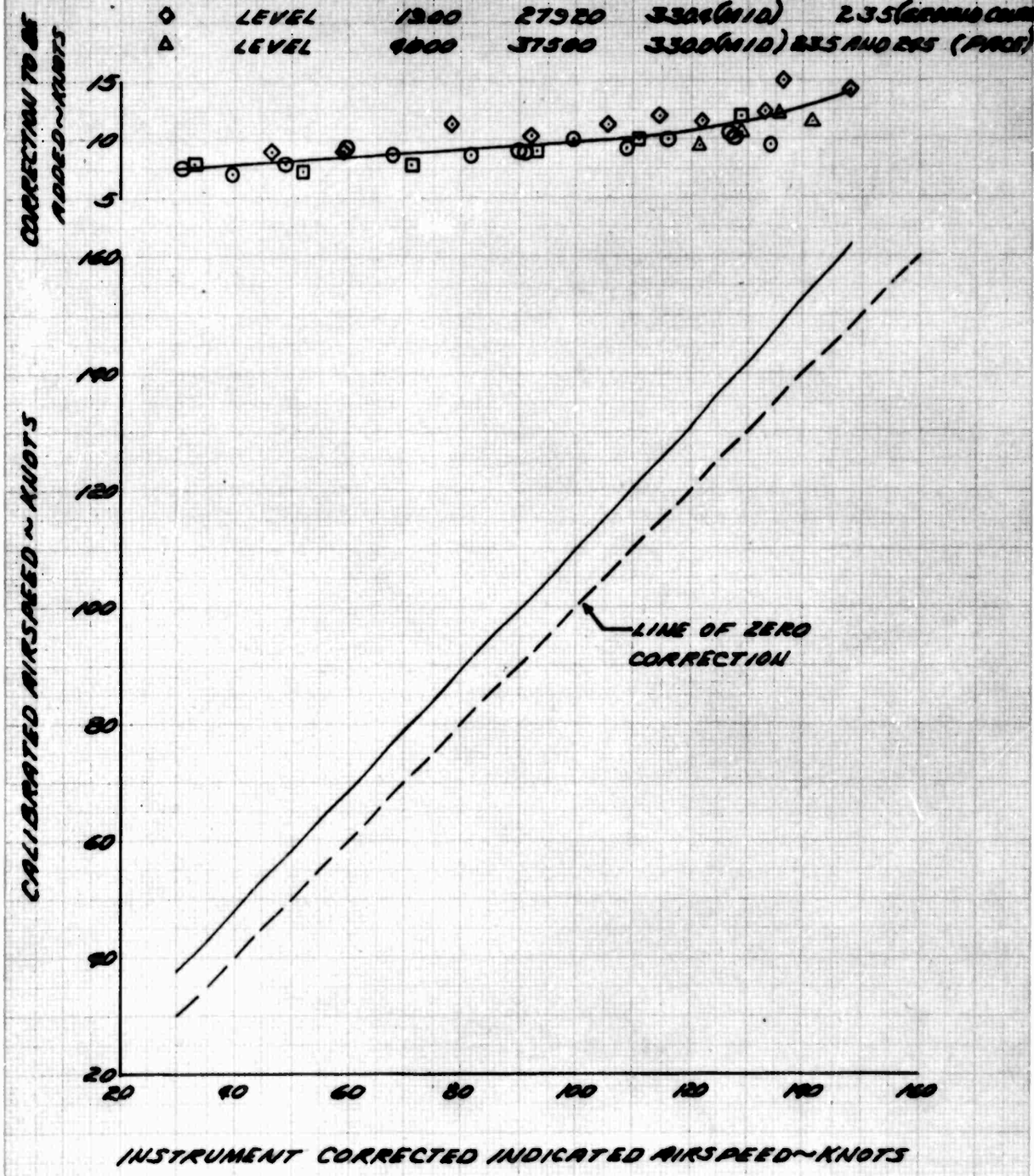


FIGURE 80
AIRSPEED CALIBRATION
CH-47C USA S/N 68-15859
STANDARD SHIPS SYSTEM
IN LEVEL FLIGHT

SYMBOL	FLIGHT CONDITION	AVG. DENSITY ALTITUDE~FT.	AVG. GROSS WEIGHT~LB	AVG. C.G. LOCATION~IN.	AVG. ROTOR SPEED~RPM
○	LEVEL	6720	29180	331.0 (MID)	235 AND 245
□	LEVEL	1900	27920	330.0 (MID)	235
◇	LEVEL	4000	31500	330.0 (MID)	235

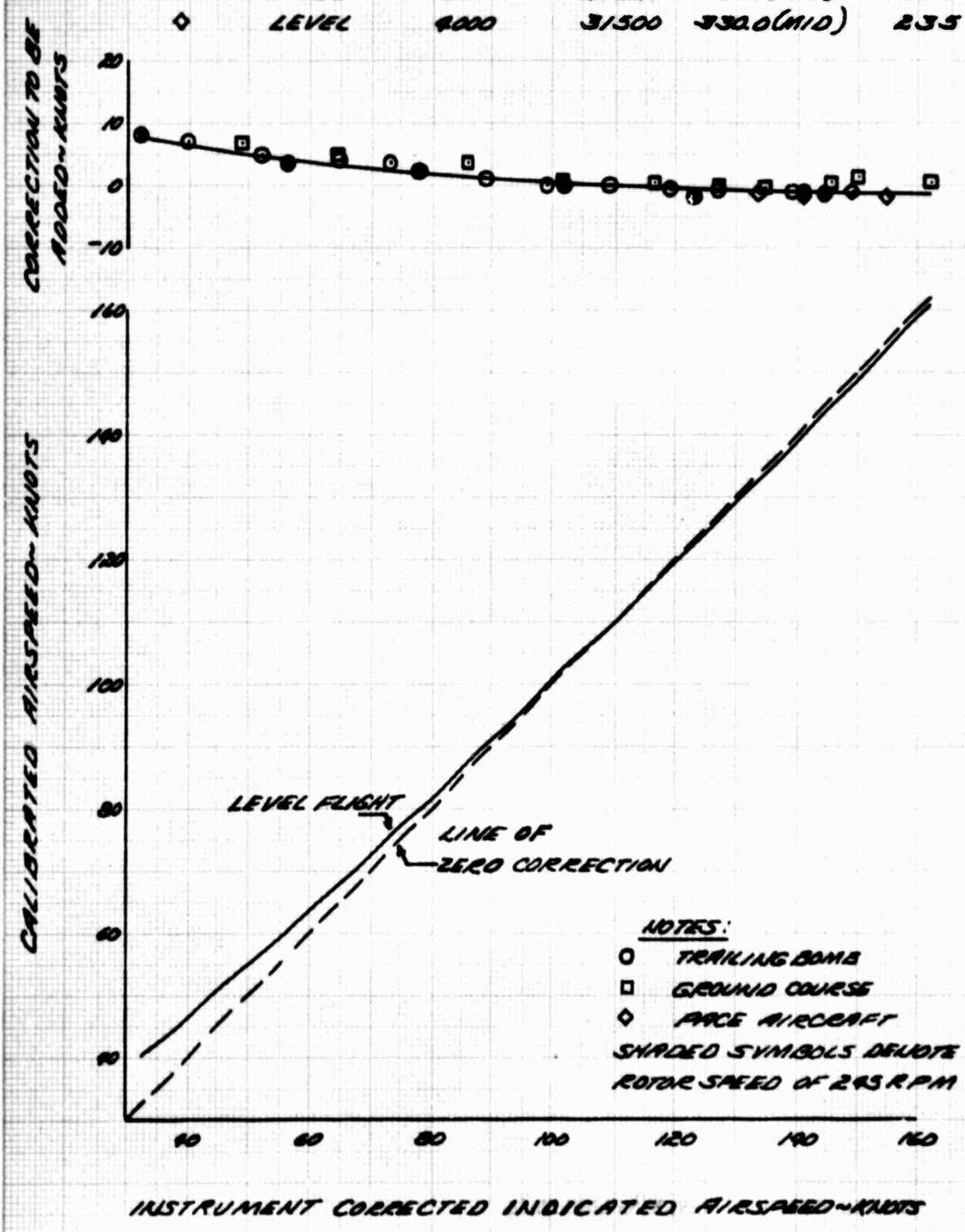
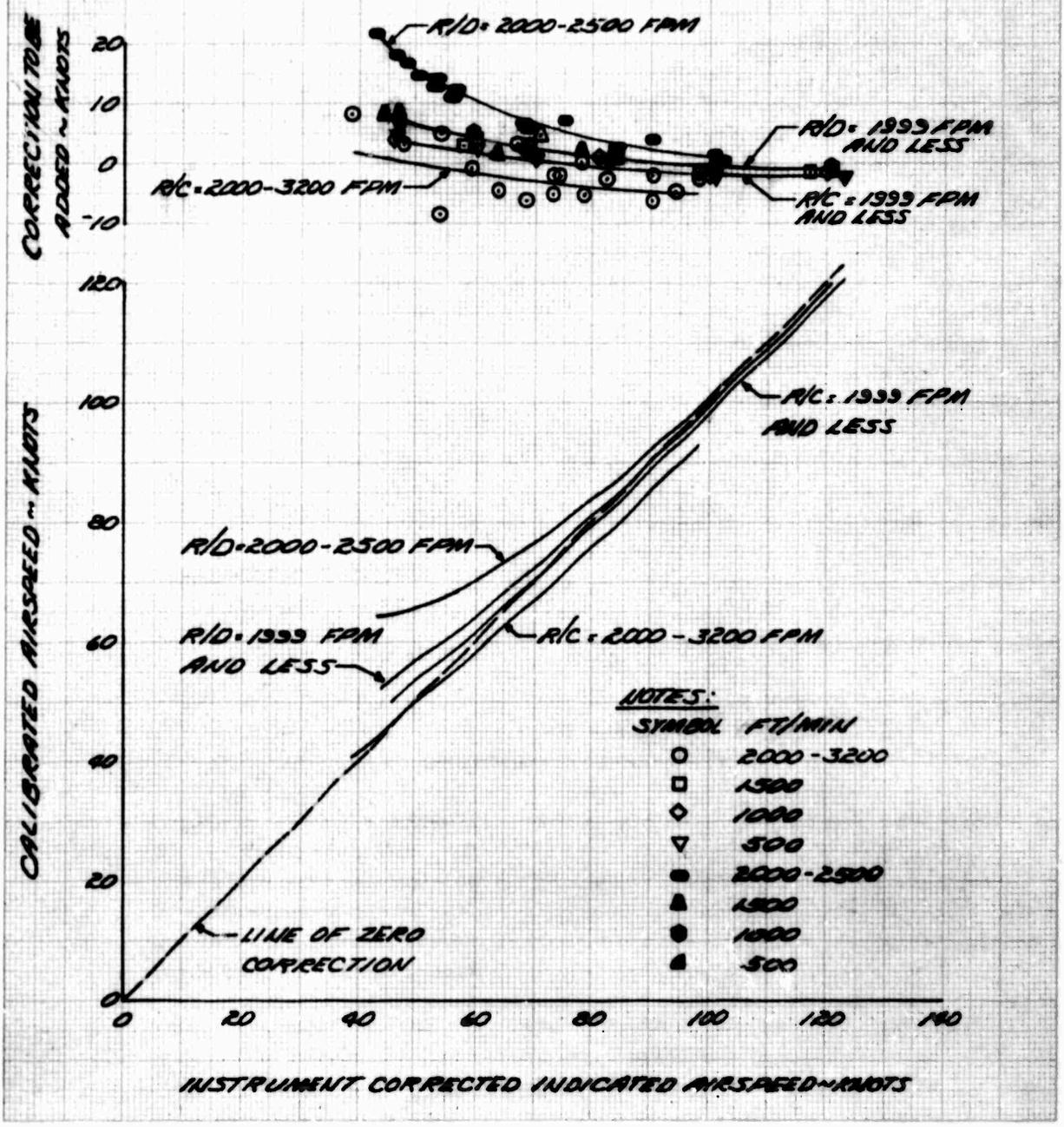


FIGURE 81
AIRSPEED CALIBRATION
CH-47C USA S/N 68-15859
STANDARD SHIRS SYSTEM
IN CLIMB AND DESCENT

SYMBOL	FLIGHT CONDITION	AVG. DENSITY ALTITUDE-FT	AVG. GROSS WEIGHT-LB	AVG. C.G. LOCATION-IN.	AVG. ROTOR SPEED-RPM
○ □ ◇ ▽	CLIMB	5000	25820	330.9 (AVG)	233
● ▲ ◆ ◢	DESCENT	5000	25820	330.9 (AVG)	230



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13. ABSTRACT

The CH-47C was flight tested to obtain detailed performance data and to verify compliance of the aircraft with the manufacturer's detail specification and applicable military specifications. The test results show that the helicopter exceeded all performance guarantees and complied with all specifications against which it was tested ~~except airspeed position errors~~. The inaccuracy of the engine torquemeter system and high engine compartment vibration levels were the only two deficiencies found. Seven shortcomings were noted for which correction is desirable: (1) objectionable cockpit vibration levels which limit maximum level-flight airspeed, (2) moderate pilot effort required to maintain optimum climb airspeeds, (3) 3/rev airspeed indicator needle oscillations at high power settings, (4) engine torque mismatch resulting from adjusting rotor speed, (5) use of landing gear power steering control may be lost at gross weights below 30,000 pounds, (6) objectionable cargo compartment vibration, and (7) objectionable noise levels in the cockpit. The small airspeed system position error associated with changes in vertical speed represent a marked improvement over the systems in the CH-47A and the CH-47B. The greatly improved hover capability and excellent climb performance enhance the operational suitability of the helicopter. The use of a cruise guide indicator to display inflight loads on the aft dynamic components of the flight control system is excellent and should be incorporated in future designs. The performance characteristics of the helicopter are satisfactory for operational use.

DD FORM 1 NOV 65 1473 REPLACES DD FORM 1473, 1 JAN 64, WHICH IS OBSOLETE FOR ARMY USE.

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14. KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
The CH-47C was flight tested						
Obtain detailed performance data						
Verify compliance						
Exceeded all performance guarantees						
Complied with all specifications except airspeed position errors						
Two deficiencies						
Seven shortcomings						
Greatly improved hover capability						
Excellent climb performance						
Use of cruise guide indicator should be incorporated						